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No. 12885

United States
Court of Appeals
for the Ninth Circuit.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION and AMERICAN AIR LINES, INC.,

Appellants,

vs.

MAURICE A. GARBELL, INC., and GARBELL RESEARCH FOUNDATION,

Appellees.

Transcript of Record

Volume III
Book of Exhibits
(Pages 605 to 834)

Appeal from the United States District Court, Southern District of California, Central Division.

Phillips & Van Orden Co., 870 Brannan Street, San Francisco, Calif.

PLAINTIFF'S EXHIBIT NO. 2
Admitted November 21, 1948.

May 18, 1948.
M. A. GARBELL
2,441,758

FLUID-FOIL LIFTING SURFACE

Filed July 16, 1946
3 Sheets-Sheet 1

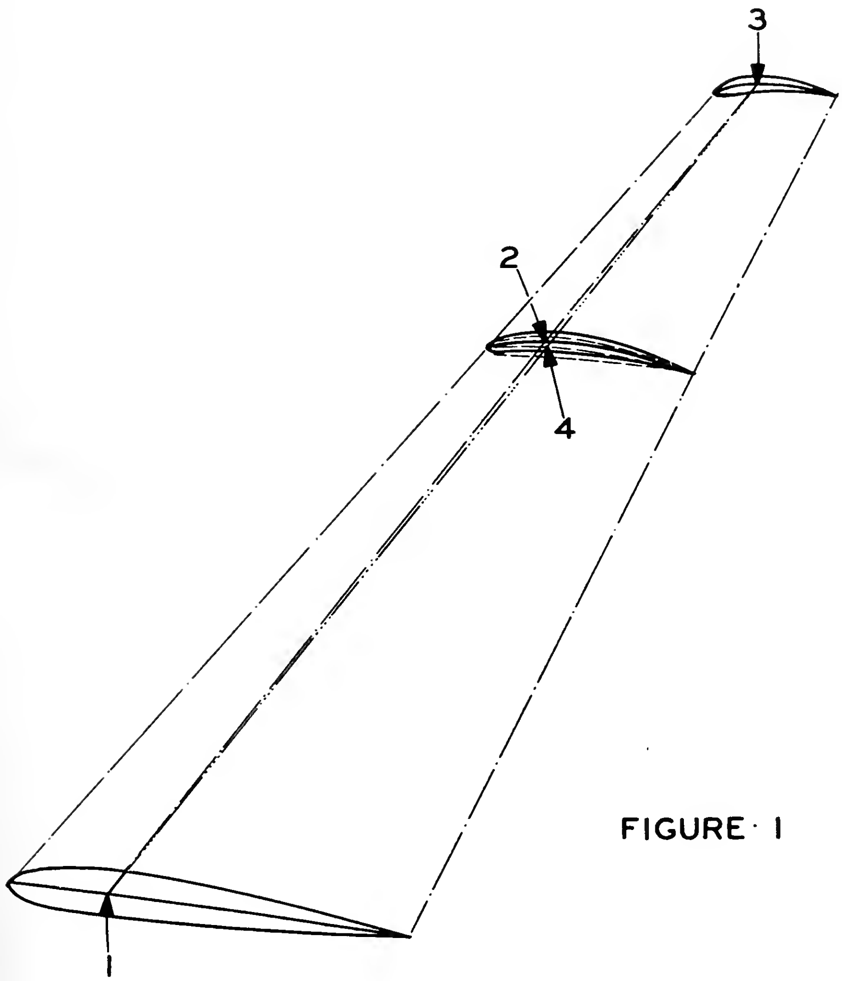


FIGURE · 1

Maurice A. Garbell INVENTOR.

BY Naylor and Lassagne
ATTORNEYS

FLUID-FOIL LIFTING SURFACE

Filed July 16, 1946

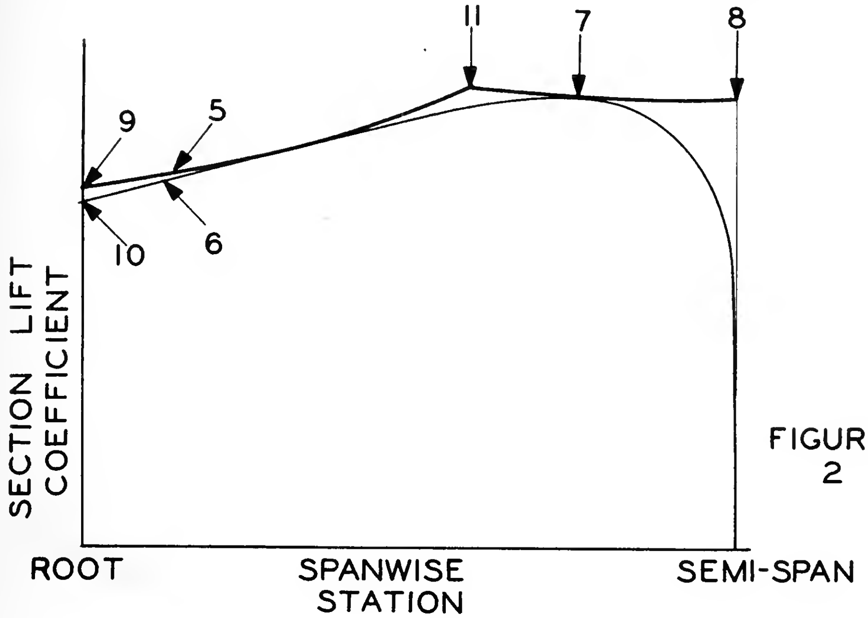


FIGURE 2

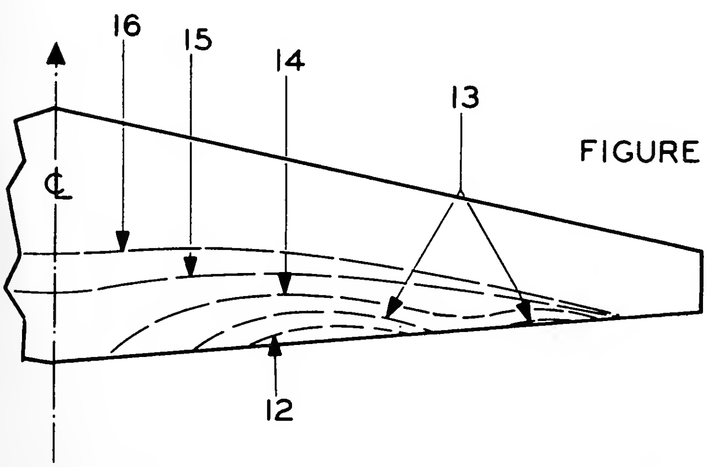


FIGURE 3

Maurice A. Garbell INVENTOR.

BY Naylor and Lassagne

ATTORNEYS

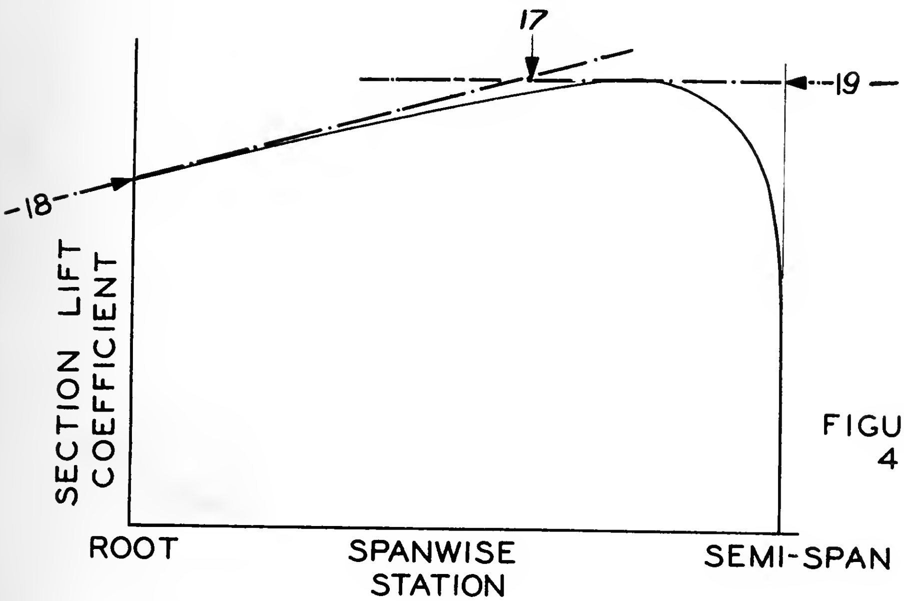


FIGURE 4

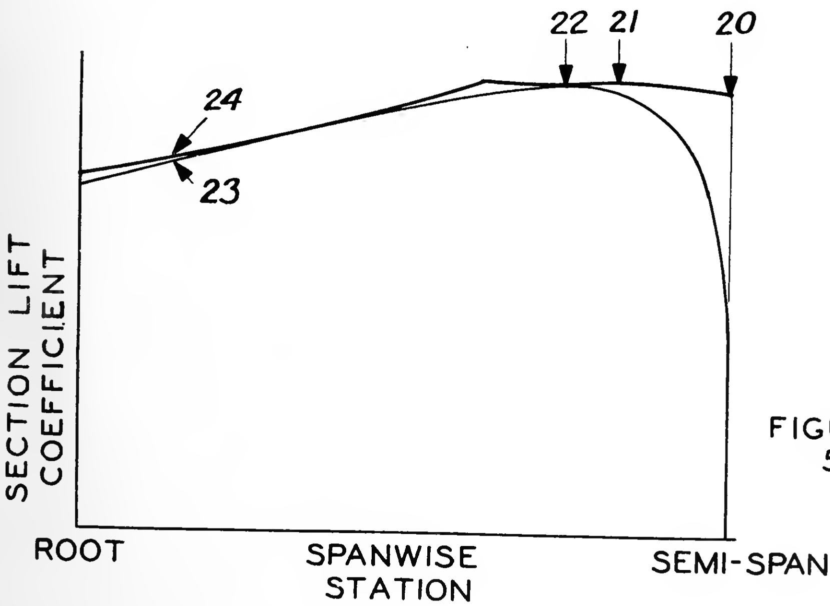


FIGURE 5

Maurice A. Garbell INVENTOR.

BY Naylor and Lassagne

ATTORNEYS

UNITED STATES PATENT OFFICE

2,441,758

FLUID-FOIL LIFTING SURFACE

Maurice Adolph Garbell, San Francisco, Calif., assignor to Maurice A. Garbell, Inc., San Francisco, Calif., a corporation of California

Application July 16, 1946, Serial No. 683,815

15 Claims. (Cl. 244—35)

This invention relates to the design and con-truction of surfaces to be driven through a fluid, ntended to produce a useful force component erpendicular to the relative velocity of the fluid vith respect to the surface, known in the art as lift force," "side force," etc., and referred to ereinafter as "lift."

In particular this invention relates to the de-ign and construction of surfaces to be driven hrough the air, intended to produce an aerody-amic lift force perpendicular to the relative vind velocity with respect to the said surface, vhile minimizing the aerodynamic drag force arallel to the relative wind. In the art such urfaces are known as "wings," "fins," "blades," tc., and will be referred to hereinafter as "lift-ng surfaces." The closed curves resulting from ntersections of the lifting surfaces with vertical lanes parallel to the relative wind will be re-erred to hereinafter as "fluid-foil sections." 'he body to which the lifting surface is fastened vill be referred to hereinafter as the "craft."

Figure 1 illustrates the preferred embodiment f this invention comprising a lifting surface de-igned and constructed according to the method utlined in the subject specification.

Figure 2 illustrates the spanwise distribution f actually prevailing section lift coefficients and he spanwise distribution of maximum attainable ection lift coefficients on a typical lifting sur-ace designed and constructed according to the ubject method of this invention.

Figure 3 illustrates the typical inception and rowth of the stall of a lifting surface designed nd constructed according to the subject method f this invention.

Figure 4 illustrates the procedure employed in he finding of the optimum spanwise location of he third controlled fluid-foil section in a lifting urface designed and constructed according to he subject method of this invention.

Figure 5 illustrates the spanwise distribution f actually prevailing section lift coefficients and he spanwise distribution of maximum attainable ection lift coefficients on a typical lifting sur-ace designed and constructed according to the ubject method of this invention, the tip section f said lifting surface having a thickness ratio maller than the optimum thickness ratio for bsolutely maximum attainable section lift co-fficient for the series of fluid-foils employed in he lifting surface.

The general object of this invention is the at-ainment of good stalling characteristics of lift-ng surfaces, said good stalling characteristics eing achieved by the employment of three or

more controlled fluid-foil sections 1, 2, and 3, selected according to the method explained in the subject specification of this invention, wherein section 2, representing the additional controlled sections interjacent between the root and the tip of the lifting surface, is at variance with the section 4 obtainable at the respective spanwise stations by means of straight-line fairing between the fluid-foil sections located at the root and the tip of the lifting surface.

Another object of this invention is the elimination of the violent rolling moments ordinarily produced by the unavoidable asymmetry of the stalling process, because the aforementioned method of fluid-foil selection suppresses the stall inception at the tip of the lifting surface and induces stall inception at a more inwardly located panel of the lifting surface, thus reducing the rolling moments acting on the craft for a given asymmetry of lift forces on the two stalled lifting surfaces.

Another object of this invention is the maintenance of adequate lateral-control effectiveness, together with the elimination of violent unstable control forces acting on control surfaces and devices attached to the trailing edge of the tip panel, during the critical stall-inception stage of the lifting surface, because the aforementioned method of fluid-foil selection induces stall inception at a more inwardly located panel of the lifting surface, so that the fluid flow over the tip panel and hence over the said control surfaces and devices remains smooth, thus maintaining effective lateral control as well as stable and smoothly varying control forces throughout the stall of the lifting surface.

Another object of this invention, through the employment of the aforementioned method of fluid-foil selection, is to reduce both the parasite drag and the induced drag of the unstalled lifting surface, and to shift the spanwise location of the "center of drag forces" of the stalled lifting surface inwardly so that the drag moment of the stalled lifting surface with respect to a vertical axis at or near the root is reduced to a value smaller than that of a lifting surface having a stall inception near the tip thereby reducing to a minimum the power required to maintain the rotation of partially or totally stalled lifting surfaces of the "rotating-wing" or "rotating-blade" type.

Additional objects of this invention will appear hereinafter.

In the art the achievement of the objects of this invention is recognized as one of the great steps in advancing safety and efficiency in aircraft design. According to accident statistics of the Civil Aeronautics Boards and other aeronautical agencies most flying accidents, especially those accidents occurring while flying in proximity of the ground, during take-off, and when landing, are caused by the stall of the lifting surface, the severity of such accidents being attributable not so much to the loss of lift directly, as indirectly to the adverse longitudinal and lateral stability characteristics, to the loss of control effectiveness, and to the violent unstable control forces produced by the stall inception near the tip of the lifting surface.

An investigation of the fundamental reasons for unsatisfactory and hazardous stalling characteristics reveals that high plan-form taper and sweep-back of the lifting surface create three principal unfavorable effects resulting in a stall inception near the tip of the lifting surface: (1) a reduction of the scale factor known in the art as "Reynolds number" in direct proportion to the decrease of chord length from the root to the tip; according to well-known experimental evidence the maximum section lift coefficient attainable with a given fluid-foil section placed in the tip panel of the lifting surface is smaller than the maximum section lift coefficient that the same section would be capable of attaining were it placed in the root panel where the chord length and hence the Reynolds number are greater; (2) a deviation from the ideal "elliptical span-load distribution" tending to increase the lift coefficients prevailing over the tip sections and to reduce the lift coefficients prevailing over the root sections at any given total lift coefficient of the lifting surface; (3) an outwardly directed spanwise fluid cross-flow, especially on the suction side of the lifting surface; this cross-flow at high lift coefficients of the lifting surface in an additional incentive for fluid-flow separation and stall near the tip of the lifting surface.

In the art, prior to this invention, it was customarily sought to counteract the aforementioned factors that contribute to the stall inception in the tip panel by resorting to the following measures: (*a*) effective washout, that is, washout of the zero-lift line of the fluid-foil section at the tip with respect to the zero-lift line of the root section, thus reducing the effective angle of attack of the tip section below the effective angle of attack of the root section; (*b*) the employment of a fluid-foil section with a more highly cambered mean line at the tip of the lifting surface than at the root, in order to enable the tip section to attain higher maximum section lift coefficients.

These measures, however, have not been entirely successful in suppressing the stall inception near the tip of the lifting surface; the spanwise distribution of the actually prevailing section lift coefficients reaches a peak near the tip and therefore inevitably intersects the nearly linear spanwise distribution of maximum attainable section lift coefficients in this most critical portion of the lifting surface.

As a rule the resulting stall patterns remain unsatisfactory for all but the lowest of plan-form taper ratios, and may become dangerously critical for plan-form taper ratios in excess of 3:1 and for any highly swept-back lifting surfaces. The stall inception in the vicinity of the tip of the lifting surface and a comparatively slow inboardward progression of the stall with any further increase of the angle of attack of the lifting surface results in the most vicious type of tip stall, with little or no stall warning, violent rolling moments, loss of lateral control, violent unstable control forces, and unstable nose-up pitching moments throughout the stall.

It was therefore customary in the art, prior to this invention, to employ as much washout and camber variations as was deemed permissible, and to transfer the further responsibility for the avoidance of the admittedly unsatisfactory stalling characteristics to the care of the pilots, or to warning signals actuated by the stalled fluid flow, or to a limitation of the elevator control travel to prevent the attainment of the high angles of attack at which stall occurs.

Techniques utilizing three controlled fluid-foil sections, in which the section at the semi-span center has either greater or smaller mean-line camber than the sections at the root and tip, have also failed to offer any substantial improvement of the dangerous tip-stall characteristics of highly tapered and/or swept-back lifting surfaces.

A preferred embodiment of this invention is described in the following specification; the broad scope of the invention is expressed in the claims concluding the instant application.

The invention consists of novel methods and combinations of methods described hereinafter, all of which contribute to produce a safe and efficient lifting surface.

Figure 1 illustrates the preferred embodiment of this invention, comprising a lifting surface with three or more "controlled" fluid-foil sections, in which the section with the least mean-line camber **1** is located at the root of the lifting surface, the section with the greatest mean-line camber **3** is located at the fluid-dynamically effective tip of the lifting surface (the actual tip fairing of the lifting surface may comprise a faired three-dimensional body without any identifiable mean-line camber, which is not of any consequence in the application of the subject invention), and one or more interjacent fluid-foil sections **2** are selected following the method outlined below, said interjacent fluid-foil sections having values of the mean-line camber at variance with the values **4** obtainable at the respective spanwise stations by means of straight-line fairing between the fluid-foil section located at the root and the fluid-foil section located at the tip of the lifting surface, provided that the respective values of the mean-line camber of the interjacent fluid-foil sections neither exceed the mean-line camber of the tip section nor fall below the mean-line camber of the root section. It shall be understood that the preceding considerations apply to all types of lifting surfaces regardless of the respective thickness ratios of the root and tip sections. It shall also be understood that additional considerations relative to the respective thickness ratios of the various controlled fluid-foil sections are presented herein for lifting surfaces wherein the thickness ratio of the root section is the greatest, and the thickness ratio of the tip section is the smallest, respectively, of any fluid-foil section employed in the lifting surface.

Figure 2 illustrates the preferred manner in which this invention, through the employment of the aforementioned method of fluid-foil selection, achieves the establishment of a curvilinear polygon **5** describing the spanwise distribution of maximum attainable section lift coefficients, said curvilinear polygon being so shaped that it envelops closely the curve **6** describing the spanwise distribution of the actually prevailing section lift coefficients, except that beyond the spanwise point 7 at which the highest actually prevailing section lift coefficient occurs the maximum attainable section lift coefficient exceeds substantially the actually prevailing section lift coefficient, so that the stall inception occurs near mid-semispan, spreads more prevalently inboardward and to a smaller extent outboardward, and does not involve the extreme tip of the lifting surface prior to the breakdown of the fluid flow over the entire remaining lifting surface.

As used herein the curvilinear polygon 5 describing the spanwise distribution of maximum attainable section lift coefficients is established by the respective values of the maximum attainable lift coefficients of the root section 9, the tip section 8, and the third or additional control section 11, and by the respective maximum attainable lift coefficients 5 of the sections obtained by conventional fairing between each pair of controlled sections 9—11, 11—8, etc.

The curve 6 describing the spanwise distribution of the actually prevailing section lift coefficients at the maximum lift coefficient of the lifting surface is obtained by conventional methods of experimentally verified calculation for the desired lifting surface, taking into consideration the plan-form, effective aerodynamic washout, section lift-curve-slope characteristics, etc.

The term "envelopment" as used herein signifies the establishment of curvilinear polygon 5 on the convex side of curve 6, wherein each individual branch 9—11, 11—8, and so forth of the curvilinear polygon 5 is tangent or nearly tangent to curve 6.

Figure 3 illustrates the stall progression resulting from the employment of the subject method of this invention. The curves 12, 13, 14, 15, and 16 indicate, in their orderly progression, the extent of the stalled lifting-surface area at angles of attack greater than the angle of attack at which stall inception 12 first occurs. This spanwise far-reaching yet gradual spread of the stalled area prevents the formation of a deep local stall in a chordwise or depthwise sense at any one spanwise station. Steep spanwise pressure differences between unstalled sections and stalled sections, and hence deep spanwise cross-flows, are thereby effectively prevented.

The prevalently inboardward development of the stalled area not only produces the desired timely stall warning in the form of a gentle tail shake at a speed slightly in excess of stalling speed, but serves also to reduce the downwash of the fluid flow aft of the lifting surface in the space usually occupied by the horizontal stabilizer, so that an upwardly directed lift-force increment is made to act on the horizontal stabilizer, thereby imposing a nose-down pitching moment on the craft that induces the craft to return to smaller angles of attack and brings to a halt any further progress and intensification of the stalling process by precluding any increase in angle of attack beyond the stalling angle.

The following specification outlines the method employed in the design of the subject lifting surface of this invention, whereby to select the most opportune values of fluid-foil section mean-line camber and fluid-foil section thickness ratio required to achieve the objects of the instant invention:

To apply the subject method of this invention it is actually necessary to know only the plan form of the lifting surface and the desired stall pattern. Inasmuch as practical considerations other than those pertaining solely to the control of the stalling characteristics ordinarily predetermine certain design parameters of the lifting surface, preferred embodiments of the subject method of this invention are hereinafter explained for two typical combinations of predetermined basic design parameters:

In the first typical configuration the following design parameters, for example, are assumed to be given a priori: (*a*) the plan form of the lifting surface, based on structural and practical design considerations; (*b*) the series of fluid-foil sections to be employed, based on high-speed and other performance requirements; (*c*) the maximum permissible effective aerodynamic washout, based on drag considerations and structural bending-moment limitations; (*d*) the thickness ratio of the fluid-foil section at the root, based on the critical-Mach-Number requirements and structural weight considerations; (*e*) the thickness ratio of the fluid-foil section at the tip, based on practical space requirements for control-surface balances, etc.; (*f*) the mean-line camber of the fluid-foil section at the tip, based on the requirement of adequate torsional lifting-surface stiffness at high speed.

The subject method of this invention is employed firstly to design the lifting surface without any effective aerodynamic washout, that is, with the three or more controlled fluid-foil sections placed at such an angle of incidence with respect to the reference chord plane of the lifting surface that the said fluid-foil sections operate at their respective zero-lift angles of attack when the entire lifting surface operates at its angle of attack for zero overall lift.

Based on fundamental experimental wind-tunnel data available for the pre-selected series of fluid-foil sections, graphs are plotted showing the variation in the maximum attainable section lift coefficient versus the mean-line camber, thickness ratio, and Reynolds number, respectively; similar graphs are plotted showing the variation in the section zero-lift angle of attack versus the mean-line camber, thickness ratio, and Reynolds number, respectively.

The approximate maximum attainable lift coefficient of the entire lifting surface for appropriate values of the Reynolds number is estimated, for example, by dividing the maximum attainable section lift coefficient of the tip section 8 (obtained from the aforementioned wind-tunnel data) by the highest spanwise value of the "additional section lift coefficient

$$C_{l_{a_1}}$$

(as defined in Army-Navy-Commerce ANC-1(1) entitled "Spanwise Air-Load Distribution"), as follows:

$$C_{L_{\max}}=\frac{C_{l_{\max\ \mathrm{tip}}}}{C_{l_{a_{1_{\mathrm{highest}}}}}}$$

this equation yields that lift coefficient of the entire lifting surface at which the most highly loaded section 7 carries a section lift coefficient substantially equal to the maximum attainable section lift coefficient 8 of the fluid-foil section employed at the tip.

The spanwise distribution 6 of the actually prevailing section lift coefficients is then calculated for the maximum lift coefficient $C_{L_{\max}}$ of the entire lifting surface, following one of the conventional calculation methods, for example, the method outlined in the Army-Navy-Commerce Manual ANC-1(1).

For the Reynolds number and the pre-selected thickness ratio of the root section, the required value of mean-line camber is determined from the graph showing the experimentally determined variation of the maximum attainable section lift coefficient with varying mean-line camber, selecting that value of the mean-line camber that produces a maximum attainable section lift coefficient 9 equal to or slightly superior to the section lift coefficient 10 actually prevailing over the root section.

For the spanwise location of the third and additional controlled sections 2 and 11, the subject method of this invention utilizes preferringly locations between the spanwise point of the highest actually prevailing section lift coefficient 7 and the root 10 of the lifting surface; the most efficient interval wherein to locate the third controlled section lies between the spanwise point of the highest actually prevailing section lift coefficient 7 and the spanwise point located twice as distantly from the tip as point 7, with a preferable optimum at the point 17, where the tangent to the inboard portion of the curve of spanwise distribution of the actually prevailing section lift coefficients 18 intersects the horizontal tangent 19 to the same curve, as shown in Figure 4.

It will be understood, however, that inescapable practical design considerations may require that the additional controlled sections 2 and 11 be placed at spanwise stations located inside power plant nacelles or at those spanwise stations where the lifting surface is mechanically jointed for sudden changes in plan-form taper, or sweepback, as is the case in craft with removable or foldable outboard panels.

The Reynolds number is calculated for the third controlled section; the thickness ratio obtainable at the third section by straight-line interpolation between the root section and the tip section is also determined. For the Reynolds number and thickness ratio thus determined, the required value of mean-line camber is found from the graph showing the experimentally determined variation of the maximum attainable section lift coefficient with varying mean-line camber, selecting that value of the mean-line camber which produces a maximum attainable section lift coefficient 11 and 17 equal to or slightly superior to the highest actually prevailing section lift coefficient 7.

From the foregoing, it will be readily seen that the lifting surface obtained by the invention, and defined by the curvilinear polygon 5, embodies the combination of an airfoil section 1 or 9 having the smallest mean line camber at the root, an airfoil section 3 or 8 having the greatest mean line camber at the tip, and one or more interjacent controlled sections 2 or 11, having values of the mean line camber at variance with the values 4 obtainable at the respective spanwise stations by means of straight line fairing between the root section and the tip section.

If the required maximum attainable section lift coefficient for the interjacent section 11 cannot be obtained with a mean-line camber not exceeding the mean-line camber of the tip section, a value equal to or slightly less than the mean-line camber of the tip section is selected. The maximum attainable section lift coefficient of the interjacent section is then increased by changing the section thickness ratio in the proper sense, usually downward, until either the required

maximum attainable section lift coefficient 11 is obtained, or until structural considerations interfere with the continuance of this procedure. If this process does not offer a conclusive result, which is rare, a small amount of effective aerodynamic washout is then introduced, ½° to 1° in each step of the application of the method, wherein the total effective aerodynamic washout is distributed in appropriate fashion between the controlled sections and where the total washout is less than the maximum permissible washout as defined in the aforelisted initial design assumptions. The entire heretofore specified procedure including the establishment of a curve 6 conforming to the washout chosen, is then repeated for the selected amount of effective aerodynamic washout, until the desired results as illustrated in Figures 2 and 3 are attained.

A typical example of the application of the principles of this invention to one well-known type of lifting surface is as follows: Here we assume a planform taper ratio of three to one, an aspect ratio of ten, a total effective aerodynamic washout of zero degrees, a constant section thickness ratio of twelve per cent along the entire semi-span, the utilization of "64—" series NACA "low-drag" fluid-foil sections, a mean-line camber of the root section 1 characterized by an "ideal lift coefficient" C_{l_i} equal to 0.1, and a mean-line camber of the tip section 3 characterized by an "ideal lift coefficient" C_{l_i} equal to 0.45. The term "ideal lift coefficient" is to be interpreted as defined by the National Advisory Committee for Aeronautics nomenclature and is herein used as a parameter characteristic of the mean line camber of a fluid foil section. Calculations based on conventional methods will indicate that a lifting surface having the above general design parameters will experience, at its maximum resultant lift coefficient, a distribution of section lift coefficients as illustrated in curve 6.

Following the procedures hereinbefore described, we achieve in the above-outlined construction the desirable stalling characteristics taught by this invention through the use of a controlled fluid-foil section 2 or 11 at a station approximately 55 per cent of the semi-span from the root and with an effective aerodynamic washout of zero degrees with respect to the root section, wherein the mean-line camber of the interjacent controlled section 2 or 11 is characterized by an "ideal lift coefficient" C_{l_i} equal to 0.35. In this structural example the mean-line camber of the interjacent controlled section 2 or 11 is greater than that of the root section 1 or 9, smaller than that of the tip section 3 or 8, and greater than that of the interpolated section 4 obtainable at the 55-per-cent semi-span station by means of straight-line fairing between sections 1 and 3, and which accomplishes the envelopment of curve 6 by the curvilinear polygon 5.

In another typical example, a lifting surface is assumed as having substantially identical basic design geometry as the preceding example, except for a structurally desirable root thickness ratio of twenty-three per cent, a tip thickness ratio of seven per cent, a total effective aerodynamic washout of one degree, and a thickness ratio of fifteen per cent at an interjacent station located at approximately 60 per cent of the semi-span.

Again following the procedure of this invention we achieve in the abovedescribed construction the desirable stalling characteristics taught by this invention through the use of a controlled fluid-foil section 2 or 11 at the station located approximately 60 per cent of the semi-span from the root and with an effective aerodynamic washout of 0.5 degree with respect to the root section, wherein the mean-line camber of the interjacent controlled section 2 or 11 is characterized by an "ideal lift coefficient" C_{l_i} equal to 0.12. In this structural example the mean-line camber of the interjacent controlled section 2 or 11 is greater than that of the root section 1 or 9, smaller than that of the tip section 3 or 8, and smaller than that of the interpolated section 4 obtainable at the 60-per-cent semi-span station by means of straight-line fairing between sections 1 and 3, and which accomplishes the envelopment of curve 6 by the curvilinear polygon 5.

(2) The second typical configuration differs from the first in that the thickness ratio of the tip section 3 is not predetermined. Hence, the following design parameters are assumed to be given a priori: (*a*) the plan form of the lifting surface; (*b*) the series of fluid-foil sections to be employed and their fluid-dynamic characteristics; (*c*) the maximum permissible effective aerodynamic washout; (*d*) the thickness ratio of the fluid-foil section at the root; (*e*) the mean-line camber of the fluid-foil section at the tip.

In this case where the thickness ratio of the tip section is not predetermined but is left to the judgment of the fluid-dynamical design engineer, the subject method of this invention employs to good advantage a peculiarity observed in the variation of the maximum attainable section lift coefficient with varying section thickness ratio. Most series of related fluid-foil sections reach their absolutely highest maximum section lift coefficient (for a given mean-line camber and Reynolds number) at a certain experimentally determined thickness ratio, usually between 12% and 16%. Sections with thickness ratios greater or smaller than optimum attain less than the absolutely maximum section lift coefficient. If, as illustrated in Figure 5, a thickness ratio smaller than optimum is used at the tip 20 of a lifting surface, where the actually prevailing section lift coefficients are greatly below their highest spanwise value 22, the fluid-foil section with the optimum thickness ratio can be located at a spanwise station 21 a small distance inboard of the tip, near the spanwise station 22 at which the highest actually prevailing section lift coefficient is encountered. Here it will be understood that the mean-line camber of the interjacent controlled section 2 may be greater or smaller than that of the aforementioned section 4, depending on the range of section thickness ratios encountered between the root and the tip of the lifting surface.

In this case the subject method of this invention is modified to the extent that, in calculating the spanwise distribution of the actually prevailing section lift coefficients 23, the maximum lift coefficient $C_{L_{max}}$ of the entire lifting surface shall be determined not on the basis of the maximum attainable section lift coefficient of the tip section, but on the basis of the absolutely maximum attainable section lift coefficient 21, that is, for the section of optimum thickness ratio, as follows:

$$C_{L_{max}}=\frac{C_{l_{max\ abs.}}}{C_{l_{a_{l_{highest}}}}}$$

The thickness ratio of the fluid-foil section at the tip of the lifting surface is then so chosen that the section 21 with optimum thickness ratio for absolutely maximum attainable section lift coefficient lies between the spanwise station of highest actually prevailing section lift coefficient 22 and the tip 20, unless structural and other design criteria interfere by establishing a minimum section thickness ratio.

If the designer intends to achieve positive stall inception in a certain spanwise panel of the lifting surface, the subject method of this invention provides that in either of the aforedescribed design procedures the mean-line camber and thickness ratios, as well as the spanwise location, of the sections comprised within or adjacent to the panel for which stall inception is desired be so selected that within the "stall inception panel" the curve of maximum attainable section lift coefficients lies slightly below the curve of actually prevailing section lift coefficients, without modifying the aforedescribed relationship of the maximum attainable section lift coefficients and the actually prevailing section lift coefficients on the remainder of the semispan of the lifting surface outside of the "stall-inception panel" proper.

If, in any of the aforedescribed cases, the lifting surface under consideration is modified by excrescences such as, for example, power-plant nacelles, or flaps that modify the local zero-lift angle and the local maximum attainable section lift coefficient, the calculation of the spanwise distribution of the effective washout and the maximum attainable section lift coefficients takes due account of the effects of these modifications by introducing "equivalent values" of the effective washout and section mean-line camber into the subject method of this invention.

Upon completion of the procedure outlined for the subject method of this invention, the zero-lift angles of the fluid-foil sections selected thusly are determined for their respective mean-line cambers, thickness ratios, and Reynolds numbers, and each fluid-foil section is set properly with respect to the reference chord plane of the lifting surface, so that the desired effective washout is achieved.

By practicing my invention a lifting surface can be designed and constructed to achieve the objects heretofore stated.

Numerous flight tests and wind-tunnel tests in reputable wind-tunnels such as the California Institute of Technology, the Massachusetts Institute of Technology, the various wind tunnels of the National Advisory Committee for Aeronautics, and elsewhere have demonstrated convincingly that each of the objects of this invention has been fully achieved. The tests were performed on numerous wing models, on sailplanes, and on models of at least five aircraft designs of widely varying design scope employing a wide variety of airfoil series. Force-test records, photographic records, and cinematographic records of the tests substantiate the attainment of the objects of this invention.

The inventor wishes it to be clearly understood that the greatly improved and generally judged satisfactory stalling characteristics of the wings (and other lifting surfaces) designed and constructed according to the subject method of this invention are directly attributable to the use of three (or more) controlled fluid-foil sections selected according to the hereinbefore specified method of this invention, and to the aforedescribed method employed in the design of such lifting surfaces.

This invention accomplishes an important improvement in the art, and the discoveries herein disclosed are of great value to all types of aircraft (as well as to craft operating in other fluids), throughout their entire operating range, and especially in the critical low-speed operation where steadiness of lift and lift variation, stability of the craft, control effectiveness, and smoothness and stability of control forces are of vital importance for the safety and efficiency of the craft; also in violent maneuvers at high speeds when high lifting-surface lift coefficients comparable with those occurring at the low-speed stall are encountered and even temporarily surpassed.

I claim:

1. A lifting surface with three or more controlled fluid-foil sections, in which the first section with the smallest mean-line camber is located at the root, the second section with the greatest mean-line camber is located at the fluid-dynamically effective tip, and the third or additional fluid-foil sections are located at stations interjacent between the root and the tip, wherein the values of the mean-line camber of the interjacent fluid-foil sections are greater than the values of the mean-line camber obtainable at the respective spanwise stations by means of straight-line fairing between the fluid-foil section located at the root of the lifting surface and the fluid-foil section located at the tip of the lifting surface.

2. A lifting surface with three or more controlled fluid-foil sections, in which the first section with the smallest mean-line camber is located at the root, the second section with the greatest mean-line camber is located at the fluid-dynamically effective tip, and the third or additional fluid-foil sections are located at stations interjacent between the root and the tip, wherein the values of the mean-line camber of the interjacent fluid-foil sections are at variance with the values of the mean-line camber obtainable at the respective spanwise stations by means of straight-line fairing between the fluid-foil section located at the root of the lifting surface and the fluid-foil section located at the tip of the lifting surface, said three or more controlled fluid-foil sections having values of the mean-line camber selected in such manner that the resulting spanwise distribution of maximum attainable section lift coefficients of the three or more controlled sections forms a curvilinear polygon enveloping a curve representing the spanwise distribution of section lift coefficients for a given planform actually prevailing at the maximum attainable lift coefficient of the lifting surface.

3. A lifting surface with three or more controlled fluid-foil sections, adapted to provide stall inception within a predetermined interval of spanwise stations in which the first section with the smallest mean-line camber is located at the root, the second section with the greatest mean-line camber is located at the fluid-dynamically effective tip, and the third or additional fluid-foil sections are located at stations interjacent between the root and the tip, wherein the values of the mean-line camber of the interjacent fluid-foil sections are at variance with the values of the mean-line camber obtainable at the respective spanwise stations by means of straight-line fairing between the fluid-foil section located at the root of the lifting surface and the fluid-foil section located at the tip of the lifting surface, said three or more controlled fluid-foil sections having values of the mean-line camber selected in such manner that the resulting spanwise distribution of maximum attainable section lift coefficients of the three or more controlled sections forms a curvilinear polygon enveloping a curve representing the spanwise distribution of section lift coefficients actually prevailing at the maximum attainable lift coefficient of the lifting surface, and that the said resulting spanwise distribution of maximum attainable section lift coefficients for a given planform be so shaped that the first intersection with the spanwise distribution of actually prevailing section lift coefficients occurs in that interval of spanwise stations for which stall inception is to be obtained.

4. A lifting surface with three or more controlled fluid-foil sections, in which the first section with the smallest mean-line camber and greatest thickness ratio is located at the root, the second section with the greatest mean-line camber and smallest thickness ratio is located at the fluid-dynamically effective tip, and the third or additional fluid-foil sections are located at stations interjacent between the root and the tip, wherein the values of the thickness ratio of the interjacent fluid-foil sections are greater than the values of the thickness ratio obtainable at the respective spanwise stations by means of straight-line fairing between the fluid-foil section located at the root of the lifting surface and the fluid-foil section located at the tip of the lifting surface.

5. A lifting surface with three or more controlled fluid-foil sections, in which the first section with the smallest mean-line camber and greatest thickness ratio is located at the root, the second section with the greatest mean-line camber and smallest thickness ratio is located at the fluid-dynamically effective tip, and the third or additional fluid-foil sections are located at stations interjacent between the root and the tip, wherein the values of the thickness ratio of the interjacent fluid-foil sections are at variance with the values of the thickness ratio obtainable at the respective spanwise stations by means of straight-line fairing between the fluid-foil section located at the root of the lifting surface and the fluid-foil section located at the tip of the lifting surface, said three or more controlled fluid-foil sections having values of the thickness ratio selected in such manner that the resulting spanwise distribution of maximum attainable section lift coefficients of the three or more controlled sections forms a curvilinear polygon enveloping a curve representing the spanwise distribution of section lift coefficients for a given planform actually prevailing at the maximum attainable lift coefficient of the lifting surface.

6. A lifting surface with three or more controlled fluid-foil sections adapted to provide stall inception within a predetermined interval of spanwise stations, in which the first section with the smallest mean-line camber and greatest thickness ratio is located at the root, the second section with the greatest mean-line camber and smallest thickness ratio is located at the fluid-dynamically effective tip, and the third or additional fluid-foil sections are located at stations interjacent between the root and the tip, wherein the values of the thickness ratio of the interjacent fluid-foil sections are at variance with the values of the thickness ratio obtainable at the respective spanwise stations by means of straight-line fairing between the fluid-foil section located at the root of the lifting surface and the fluid-

foil section located at the tip of the lifting surface, said three or more controlled fluid-foil sections having values of the thickness ratio selected in such manner that the resulting spanwise distribution of maximum attainable section lift coefficients of the three or more controlled sections forms a curvilinear polygon enveloping a curve representing the spanwise distribution of section lift coefficients actually prevailing at the maximum attainable lift coefficient of the lifting surface, and that the said resulting spanwise distribution of maximum attainable section lift coefficients for a given planform be so shaped that the first intersection with the spanwise distribution of actually prevailing section lift coefficients occurs in that interval of spanwise stations for which stall inception is to be obtained.

7. A lifting surface with three or more controlled fluid-foil sections, in which the first section with the smallest mean-line camber is located at the root, the second section with the greatest mean-line camber is located at the fluid-dynamically effective tip, and one of the interjacent fluid-foil sections is located near a spanwise point where a tangent to the inboard portion of a curve representing the spanwise distribution of actually prevailing section lift coefficients for a given planform intersects a substantially horizontal tangent to the highest point of the same curve, wherein the values of the mean-line camber of the interjacent fluid-foil sections are greater than the values of the mean-line camber obtainable at the respective spanwise stations by means of straight-line fairing between the fluid-foil section located at the root of the lifting surface and the fluid-foil section located at the tip of the lifting surface.

8. A lifting surface with three or more controlled fluid-foil sections, in which the first section with the smallest mean-line camber and greatest thickness ratio is located at the root, the second section with the greatest mean-like camber and smallest thickness ratio is located at the fluid-dynamically effective tip, and one of the interjacent fluid-foil sections is located near a spanwise point where a tangent to the inboard portion of a curve representing the spanwise distribution of actually prevailing section lift coefficients for a given planform intersects a substantially horizontal tangent to the highest point of the same curve, wherein the values of the thickness ratio of the interjacent fluid-foil sections are greater than the values of the thickness ratio obtainable at the respective spanwise stations by means of straight-line fairing between the fluid-foil section located at the root of the lifting surface and the fluid-foil section located at the tip of the lifting surface.

9. A lifting surface with three or more controlled fluid-foil sections and having a highest actually prevailing section lift coefficient at a predetermined spanwise station, in which the first section with the smallest mean-line camber and greatest thickness ratio is located at the root, the second section with the greatest mean-line camber and smallest thickness ratio is located at the fluid-dynamically effective tip, and the third or additional fluid-foil sections are located at stations interjacent between the root and the tip, wherein the values of the mean-line camber of the interjacent fluid-foil sections are at variance with the values of the mean-line camber obtainable at the respective spanwise stations by means of straight-line fairing between the fluid-foil section located at the root of the lifting surface and the fluid-foil section located at the tip of the lifting surface, and wherein the aforesaid fluid-foil section at the tip of the lifting surface has a thickness ratio smaller than the optimum thickness ratio for absolutely maximum attainable section lift coefficient of the fluid-foil series employed, so that a fluid-foil section having the optimum thickness ratio obtained by conventional interpolation between two of the controlled sections lies a short distance inboard of the tip of the lifting surface, near the spanwise station at which the highest actually prevailing section lift coefficient occurs.

10. A lifting surface with three or more controlled fluid-foil sections and having a highest actually prevailing section lift coefficient at a predetermined spanwise station, in which the first section with the smallest mean-like camber and greatest thickness ratio is located at the root, the second section with the greatest mean-line camber and smallest thickness ratio is located at the fluid-dynamically effective tip, and third or additional fluid-foil sections are located at stations interjacent between the root and the tip, wherein the values of the thickness ratio of the interjacent fluid-foil sections are greater than the values of the thickness ratio obtainable at the respective spanwise stations by means of straight-line fairing between the fluid-foil section located at the root of the lifting surface and the fluid-foil section located at the tip of the lifting surface, and wherein the aforesaid fluid-foil section at the tip of the lifting surface has a thickness ratio smaller than the optimum thickness ratio for absolutely maximum attainable section lift coefficient of the fluid-foil series employed, so that a fluid-foil section having the optimum thickness ratio obtained by conventional interpolation between two of the controlled sections lies a short distance inboard of the tip of the lifting surface, near the spanwise station at which the highest actually prevailing section lift coefficient occurs.

11. A lifting surface with three or more controlled fluid-foil sections, in which the first section with the smallest mean-line camber is located at the root, the second section with the greatest mean-line camber is located at the fluid-dynamically effective tip, and the third or additional fluid-foil sections are located at stations interjacent between the root and the tip, wherein the values of the mean-line camber of the interjacent fluid-foil sections are smaller than the values of the mean-line camber obtainable at the respective spanwise stations by means of straight-line fairing between the fluid-foil section located at the root of the lifting surface and the fluid-foil section located at the tip of the lifting surface.

12. A lifting surface with three or more controlled fluid-foil sections, in which the first section with the smallest mean-line camber and greatest thickness ratio is located at the root, the second section with the greatest mean-line camber and smallest thickness ratio is located at the fluid-dynamically effective tip, and the third or additional fluid-foil sections are located at stations interjacent between the root and the tip, wherein the values of the thickness ratio of the interjacent fluid-foil sections are smaller than the values of the thickness ratio obtainable at the respective spanwise stations by means of straight-line fairing between the fluid-foil section located at the root of the lifting surface and the fluid-foil section located at the tip of the lifting surface.

13. A lifting surface with three or more con-

trolled fluid-foil sections, in which the first section with the smallest mean-line camber is located at the root, the second section with the greatest mean-line camber is located at the fluid-dynamically effective tip, and one of the interjacent fluid-foil sections is located near a spanwise point where a tangent to the inboard portion of a curve representing the spanwise distribution of actually prevailing section lift coefficients for a given planform intersects a substantially horizontal tangent to the highest point of the same curve, wherein the values of the mean-line camber of the interjacent fluid-foil sections are smaller than the values of the mean-line camber obtainable at the respective spanwise stations by means of straight-line fairing between the fluid-foil section located at the root of the lifting surface and the fluid-foil section located at the tip of the lifting surface.

14. A lifting surface with three or more controlled fluid-foil sections, in which the first section with the smallest mean-line camber is located at the root, the second section with the greatest mean-line camber is located at the fluid-dynamically effective tip, and one of the interjacent fluid-foil sections is located near a spanwise point where a tangent to the inboard portion of a curve representing the spanwise distribution of actually prevailing section lift coefficients for a given planform intersects a substantially horizontal tangent to the highest point of the same curve, wherein the values of the thickness ratio of the interjacent fluid-foil sections are smaller than the values of the thickness ratio obtainable at the respective spanwise stations by means of straight-line fairing between the fluid-foil section located at the root of the lifting surface and the fluid-foil section located at the tip of the lifting surface.

15. A lifting surface with three or more controlled fluid-foil sections and having a highest actually prevailing section lift coefficient at a predetermined spanwise station, in which the first section with the smallest mean-line camber is located at the root, the second section with the greatest mean-line camber is located at the fluid-dynamically effective tip, and the third or additional fluid-foil sections are located at stations interjacent between the root and the tip, wherein the values of the thickness ratio of the interjacent fluid-foil sections are smaller than the values of the thickness ratio obtainable at the respective spanwise stations by means of straight-line fairing between the fluid-foil section located at the root of the lifting surface and the fluid-foil section located at the tip of the lifting surface, and wherein the aforesaid fluid-foil section at the tip of the lifting surface has a thickness ratio smaller than the optimum thickness ratio for absolutely maximum attainable section lift coefficient of the fluid-foil series employed, so that a fluid-foil section having the optimum thickness ratio obtained by conventional interpolation between two of the controlled sections lies a short distance inboard of the tip of the lifting surface, near the spanwise station at which the highest actually prevailing section lift coefficient occurs.

MAURICE ADOLPH GARBELL.

REFERENCES CITED

The following references are of record in the file of this patent:

UNITED STATES PATENTS

Number	Name	Date
1,547,644	Cronstedt	July 28, 1925
1,817,275	Soldenhoff	Aug. 4, 1931
1,839,349	Sigrist	Jan. 5, 1932
1,890,079	Focke	Dec. 6, 1932

PLAINTIFFS' EXHIBIT No. 12

[Western Union Message]

BY16 113 NT. Miami, FLO., Jul 20

F. H. Fleet, President Consair

Can You Offer Advanced Field of Activity of Experienced Aeronautical Engineer. Well Versed in Airplane and Engine Design, Performance Analysis and Research. Have Three Successful Original Designs to My Credit. For the Past Three Years Have Taught Applied Mechanics. Strength of Materials, Mechanism, Advanced Structural Design, Aerodynamics, Aeronautical Meterology in Leading Aeronautical Engineering School West Coast and University of California. Am at Present Concluding Training Program in Transatlantic Airlines School Here. Two Original Technical Text Books Just Coming Off Presses New York Publisher as Well as Many Articles Published in Leading News-Papers and Magazines. Perfect Knowledge All Important European Languages Including Russian. Wire if Interested to Forestall Acceptance Other Offer. 1801 Southwest 23 Terr., Miami.

DR. MAURICE A. GARBELL.

1801 23.
1114A

Admitted November 21, 1950.

PLAINTIFFS' EXHIBIT No. 13

Western Union
[Telegram]

July 21, 1942.

Dr. Maurice A. Garbell,
1801 Southwest 23 Terr.
Miami, Florida.

Reference Your Telegram to R. H. Fleet We Have Openings in Structures Preliminary Design and Aerodynamics for Aeronautical Engineers. We Are Interested in Knowing the Kind of Connection You Are Seeking, the Salary Expected and Whether or not You Are an American Born Citizen.

B. W. SHEAHAN,
Consolidated Aircraft Corporation.

cc: Employment Dept.
Engr. File

Admitted November 21, 1950.

PLAINTIFFS' EXHIBIT No. 14

Maurice A. Garbell, D. Sc.

1801 SW 23rd Terrace,
Miami, Florida,
July 25, 1942.

Consolidated Aircraft Corporation,
San Diego, California.

Attention: Mr. B. W. Sheahan.

Gentlemen:

I have for acknowledgment receipt of your telegram of July 21st reading as follows:

> "Reference your telegram to R. H. Fleet we have openings in structures, preliminary design, and aerodynamics for aeronautical engineers. We are interested in knowing the kind of connection you are seeking, the salary expected, and whether or not you are an American born citizen."

I am primarily interested in being placed where my ability may find its greatest usefulness in your organization, namely preliminary design or research engineering.

In order that you may gain some idea of actual accomplishments, I respectfully refer you to "Aviation," June, 1939, the lead article, showing photograph of "Arcore," one of three successful designs of which a series of fifty ships were built by me. To

Plaintiffs' Exhibit No. 14—(Continued)

summarize the technical value of such advanced developments for power plane design, I might mention that all these ships had aspect ratio greater than 15, single spar wings, and monococque fuselages, stressed for aerobatics and thunderstorm soaring conditions. My planes were fitted with self-stabilizing wing-spoilers for emergency dives, zero-yaw differential aileron control and other improvements intended for added safety, maneuverability, and ease of assembly. These planes were designed, built, and successfully flown by a research institute for motorless flight under my direction, guidance, and supervision.

It is needless for me to digress further into the detailed value of applying these principles to power planes; Consolidated's adoption and development of the Davis wing, for example, indicates your recognition of their importance.

Incidentally, in connection with citizenship, I might mention that the United States Government granted me full citizenship through spontaneous and urgent intervention of the Office of the Chief of Staff, Army Air Corps, War Department, Washington, D. C., with the Naturalization Bureau after a rigid and thorough investigation. The recommendation was the result of the complete knowledge of my activities in this country and abroad by a member of the Staff Office and the recognition of my value to the present war effort:

As to salary expected, I prefer that you make an

Plaintiffs' Exhibit No. 14—(Continued)

offer to me, commensurate with the position available.

I shall look forward to your further advice, and if you are interested an early reply will be appreciated to forestall my final decision between other seemingly interesting positions offered me within the past few days.

Yours very truly,

/s/ DR. MAURICE A. GARBELL.

Maurice A. Garbell, D. Sc.

August 7, 1942.

Consolidated Aircraft Corporation,
San Diego, California.

Transcript of subjects and courses studied:

Institute of Technology, Berlin Charlottenburg:
(Technische Hochschule)

- Differential and integral calculus,
- Theory of numbers,
- General Mechanics (elementary applied mechanics and kinematics),
- Drafting,
- Descriptive geometry,
- General physics,
- General and inorganic chemistry,
- Technology of metals,
- Economy,
- History of industrial development.

Plaintiffs' Exhibit No. 14—(Continued)

Institute of Technology, Milan:
(Regio Istituto di Ingegneria & Regio Politecnico)

Differential and integral calculus (2 years.)
Analytic and projective geometry (1 yr.)
Descriptive geometry (2)
Artistic sketching (1)
Architectural drawing (1)
Engineering drawing (1)
General and experimental physics (2)
Industrial physics (general and industrial thermodynamics—1)
Analytical mechanics (1)
Applied mechanics and strength of materials (1)
Structures (1)
Science of mechanism (1 yr.)
General and inorganic chemistry (1)
Organic chemistry (audited lecture course—1)
Qualitative analytical chemistry (aud. lecture course, completed laboratory—1)
Industrial and agricultural chemistry (2)
Machine design (1)
Hydraulics (1)
Thermal and hydraulic engines (1)
Internal combustion engines (1)
Electro-engineering (1)
Building materials (1)
Metallurgy and metallography (1)
Industrial technology (1)
Topography and surveying (1)

Plaintiffs' Exhibit No. 14—(Continued)

Geology (1)
Mineralogy (1)
Industrial planning (1)
Industrial economy (1)
Transportation (1)
Appraisal of industrial plants and machinery (1)
Highway and railroad engineering (1)
Aerodynamics (1)

Thesis for doctor's degree:

a) design of a 9-cylinder 750 HP radial engine,

b) analysis of the possibilities for steam turbines on large stratosphere airplanes.

Minor theses:

a) Geology: geological survey of a certain area north of Milan, for a joint land and water airport.

b) Civil structures: a wooden hangar for a small chemical factory, and a concrete structure for a swimming pool.

c) Industrial planning: preliminary planning for a factory producing aluminum alloy cylinders for aircraft engines.

d) Aerodynamics: a report on four years of activity as a Manager of the Research In-

Plaintiff's Exhibit No. 14—(Continued)

stitute for Soaring Flight, the designs brought to completion, special projects, organization of the experimental shop, and flying activities.

/s/ MAURICE A. GARBELL.

Admitted November 21, 1950.

PLAINTIFFS' EXHIBIT No. 15

Form 182-R

Consolidated Aircraft Corporation
San Diego, California

Employment Agreement

I represent the statements made in my application for employment, submitted to Consolidated Aircraft Corporation on this date, to be correct to the best of my knowledge and belief; that no attempt has been made to conceal pertinent information; that all facts in that application are open to investigation and verification by Consolidated Aircraft Corporation; and I agree to hold Consolidated Aircraft Corporation and persons named in my application blameless should the information received from those persons result in my dismissal by Consolidated Aircraft Corporation.

I also agree to read and abide by "Laws of the United States and Proclamation of the President of the United States Relating to Classified Air Corps Projects" pertaining to espionage and sabotage which is printed on the reverse side of this sheet.

Plaintiffs' Exhibit No. 15—(Continued)

I hereby acknowledge receipt of Consolidated Aircraft Corporation's employee handbook and agree to abide by the rules and regulations set forth therein.

/s/ MAURICE A. GARBELLL.

Date: 9-7-42

Application for Employment

Consolidated Aircraft Corporation
Industrial Relations Department

Employment Division
1845 Moore Street, San Diego, California

[Stamped]: Plant Protection Sep. 8, 1942.

This Application for Employment Is Submitted to Consolidated Aircraft Corporation with Full Understanding of the Following Listed Instructions and Information:

1. Application to Be Filled Out in Own Handwriting. (Do not Print.) (Do not Type.)
2. Make Sure That Each and Every Question Has Been Answered In Full.
3. Make Sure the Employment History Section Is Complete in Detail.
4. Make Sure Your References Are Persons Who Have Known You for a Long Period of Time and Are not Relatives or Previous Employers.

Plaintiffs' Exhibit No. 15—(Continued)

5. It Is Understood That You Represent the Statements Made by You in This Application to Be Correct to the Best of Your Knowledge and Belief; That No Attempt Has Been Made to Conceal Pertinent Information; That All Facts Are Open to Investigation and Verification by Consolidated Aircraft Corporation: and Further That You Agree to Hold Consolidated Aircraft Corporation and Persons Named Herein Blameless Should Such Information Result in the Revocation of This Application, and/or Subsequent Dismissal From Employment.

6. It Is Further Understood That if Accepted for Employment, You Agree to Read and Abide by the "Laws of the United States and Proclamation of the President of the United States Relating to Classified Air Corps Projects," Pertaining to Espionage and Sabotage, a Copy of Which Is Printed in the Rules for Employees of the Consolidated Aircraft Corporation.

7. Native Born Applicants Must Present Birth Certificate or Other Official Documentary Evidence of Citizenship.

8. Foreign Born Applicants Must Present Final Citizenship Papers.

Date: August 7, 1942.

Name in Full: (Print) (Last) Garbell, (First) Maurice, (Middle) Adolph.

Social Security No.: 062-14-8883.

Plaintiffs' Exhibit No. 15—(Continued)

Local Address: (Street and Number) 1801 SW 23rd Terrace, (City) Miami, (State) Florida.

Phone Number: 48-1980.

Permanent Address: (Street and Number) 1714 Lake Street, (City) San Francisco, (State) California.

Phone Number: BAyview 9186.

Former Address: (Street and Number) 1106 Sherman Street, (City) Alameda, (State) California.

How Long There? Oct., 39—Nov., 40.

Former Address: (Street and Number) 3026-84th Street, (City) Jackson Heights, (State) New York.

How Long There? Feb., 39—Oct., 39.

Former Address: (Street and Number) 16 Hamburgas iela, (City) Riga-Meza Parks, (State) Latvia.

How Long There? 1933-1939.

Former Address: (Street and Number) 2 Jura Alunana iela, (City) Riga, (State) Latvia.

How Long there? Family resid. for two generations.

Date of Birth: (Month) May, (Date) 21, (Year) 1914.

Place of Birth: (City) Moscow, (State) Russia.

Nationality: Russian.

Plaintiffs' Exhibit No. 15—(Continued)

This Line to Be Filled in by Foreign Born Citizens Only:

Date of Entry: Feb. 28, 1939.

Port of Entry: New York City, N. Y.

Date of Second Papers: 5-5-42.

Where Issued? Superior Court, County of San Francisco, California. No. 5029278.

Draft Board Location: (City) Alameda, (State) California.

Draft Board No.: 62

Order No.: 728

Class: 2-B

Date of Class: 5-20-42

Please use following space for reason of your present classification: Essential in defense work.

Are you a member of National Guard or Reserves? no. If so, what? —

Give military or naval service, U. S. or other countries: none.

Have you ever used any other name? no.

If so, what? —

Have you ever been convicted of a felony? no.

If so, explain in following space: —

Male: yes. Female: —

Color: white.

Single: — Married: yes.

Divorced: — Widowed: —

Height: 5'11". Weight: 175 lbs.

Color of Hair: dark brown.

Plaintiffs' Exhibit No. 15—(Continued)

Color of Eyes: brown.
Scars, Birthmarks, etc.: none.
Live with Wife: yes.
Live with Parents: no.
Live with Relatives: no.
Live Alone: no.
Wife Work? no.
Number of Dependent Children: none.
Number of Dependent Parents: 1.
Number of Other Dependents: 1.
Own Home: no. Rent: yes.
Room: — Board: —
How long in California? Oct. '39-May, 1942.
What Counties? Alameda & San Francisco.
How long in San Diego? —
Are Dependents in San Diego? no.
If not, where? Wife with me, Mother at present in British Mandate of Palestine.
Are you going to bring them here? no (except wife)
Father's Name: Edward Garbell.
Birthplace: Goldingen, Russia.
Present Address: deceased 1919.
Mother's Name: Flora, nee Feitelberg.
Birthplace: Goldingen, Russia.
Present Address: 23 Ussishkin St., Jerusalem (Palestine).
Wife (or Husband) Esther, nee Feitelberg.
Birthplace: San Francisco, California.

Plaintiffs' Exhibit No. 15—(Continued)

Present Address: 1801 SW 23rd Terrace, Miami, Florida.

Names and Addresses of Near Relatives now residing in Foreign Countries: Mother (please refer to above address)

Names and Relationship of Relatives Employed by this Company: none.

Do you have Relatives working for other Aircraft Companies? no. Which Companies? —

List Clubs, Societies, and Fraternal Organizations of which you are a Member: Institute of the Aeronautical Sciences, American Meteorological Society, Soaring Society of America, International Research Committee for Motorless Flight.

What are your hobbies and other interests? Sailing, soaring, swimming, photography, meteorology.

CHOOL	No. Yrs.	Year Left	Graduated	Degree	Major Subjects and Courses Liked Best	NAME OF SCHOOL	City and State
le	3	1923	yes	cert. of graduation	normal 4-year course	Dr. Busch's private sch.	Heidelberg, Germany.
....	9	1932	yes	" " " "	Mathematics, Physics, Chem., Geography, History, Latin, Greek,	"Gymnasium" Heidelberg and Berlin- Zehlendorf	-Germany.
ege	6	1938 Nov. 10	yes	Doctor in Mech. & Indust. Engin'g.	Aerodynamics, Structures, Aircraft Engines. (see also attached transcript)	Institutes of Technology Berlin-Charlottenburg (1 yr.) and Milan	Italy (5 yrs.)
le or ational							

NOTICE All Applicants Showing Vocational Training must be able to furnish Transcript of School record.

e you ever filed application for employment with this company before? no When? - Where? -

ve you ever worked for this company before? no When? - Why did you leave? -

PREVIOUS EMPLOYMENT RECORD

NOTICE — Account for all Periods of Unemployment

Name of Employer and Type of Business	Department and Address	Your Position and Duties	Dates by Month and Year	Rate of Pay	Why Did You Leave?
ent or Last: eing School of ronautics	Oakland, California	In charge of aeron. engineering courses. Appl. Mech., Strength of Materials, Science of Mechanism, Advanced Structures, Basic & Advanced Aerodynamics, Basic Aeron. Meteorology.	Oct. 39 May 42		To complete technical text work under contract for publication and organize vitally needed technical training courses for transoceanic ferry airline.
t to Last					
xt Previous					
xt Previous: iversity of lifornia.	Extension Division, Berkeley, California	CPT instruction Meteo. & Nav. Elem. Meteorology	1940 - May 42		
xt Previous: I Development rporation	120 Liberty St., New York, N.Y.	Experimental development work and design on automatic engine controls for Mr. Fokker's yacht "Q.E.D."	May 39 July 42		Completion of project followed by dissolution of firm.
xt Previous					
xt Previous: omposition of technical articles (for example, "Aviation", une and August, 1939) and a technical report to the CAA.			Feb. 28, 39 to May 39		
xt Previous: ravels (France, Italy, England, U.S.)		to	Nov. 38 Feb. 39	after conferment of degree on November 10, 1938.	
xt Previous: rch Division of ne Auto-Union	Zwickau-Germany	apprentice engineer	Mar. 32 Oct. 32	learned to operate all standard machine tools in tool department, then completed apprenticeship in inspection department and experimental shop, working on rear engine racing cars.	
Previous: umerous brief periods (summer vacations) as apprentice ngineer in small machine shops and at sea (freighter).			1932 to 1938		
xt Previous					

…you ever been discharged? no If so, where and why? -

…you ever been in business for yourself? no What kind? -

…th of Shop Experience since 193… Do you read and work from blueprints? yes

…k Precision instruments with which you are familiar: Micrometers, Calipers, Surface Gauge, Vernier Gauge, Radius Gauge. All.

…you hand tools for work desired? yes

…EE REFERENCES (Other than Previous Employers and Relatives)

…e	Business	Address
John … Thor…	Project Engineer	Vega Aircraft Corp., Burbank, Cal.
Lewin B. Barringer	Glider Specialist, Off. of the Chief of Staff, US Air Corps,	War Dept., Washington, D.C.
… Irwin E. Farington	Coordinator, District 2, Calif.	Selective Serv. H'd.q.u's, Berkeley, Calif.

…any objections to working nights, Saturdays or Sundays. none

…nic Ailments or …cal Defects? none Do you have a Rupture now? no Have you ever been ruptured? no If so, cured by Surgery or Injections? - When? -

…you ever had …r Fainting Spells? no Have you ever been vaccinated? yes If so, when and what? Smallpox and ~~tetanus as a child~~

…ou wear glasses? no Cause - Is vision corrected with glasses? -

…you ever had a severe illness or operation? no If so, when and what? -

…received compensation for …ccident or disability? no If so, give Name of Company -

…? - Type of Injury? - Amount Received? -

APPLICANT DOES NOT WRITE BELOW THIS LINE

1 2 3 4	Educ 1 2 3 4	P-EM 1 2 3 4	Att 1 2 3 4	Acct 1 2 3 4	Res't 1 2 3 4
√	Tec ✓	Iden	Evas	Qual ✓	Accep ✓
	C-Gen	Sim ✓	U-Cop	Adapt	Rejt
	C-Ext	Rela	Slow	Inex	Decl
	Voc	Dis	Arg	Attl	Hold
	H-Tec	Unql	Migr	N.R.H.	
	H-Gen	Stud	Youth	D-W-J	
	1234		Unk't	W-MM	
	012			U-Dec	
	345			N.Pr.	
	678			Phy	

Referred by

Recommended by

Supplement mailed

Per instruction of

Supplement received

No.	Date	By
1.	9-4-42	[illegible]
2.		
3.		
4.		

…ARKS:

…No. …ked … Citizenship Checked

…p Insurance Amount Hospitalization Insurance Plan Photographed and Fingerprinted

…ance Beneficiary Relationship

…ne [redacted] Street and Number City State

Admitted November 21, 1950.

19

PLAINTIFFS' EXHIBIT No. 16

Consolidated Aircraft Corporation

3834

Invention Agreement

Agreement entered into by and between Consolidated Aircraft Corporation (hereinafter called the Company), and Maurice Adolph Garbell (hereinafter called Employee), Witnesseth:

In consideration of the mutual undertakings hereinafter set forth the parties hereto do hereby agree as follows:

1. The Employee agrees:

(a) To disclose promptly in writing to the Company's Patent Department or to such person as the Company may designate, all inventions and improvements heretofore or hereafter made, developed, perfected, devised or conceived by the Employee either solely or in collaboration with others during the Employee's employment by the Company, whether or not during regular working hours, and including a period of one (1) year after termination of employment, relating to aircraft or parts and the manufacture thereof, or relating in any way to aviation or to the business, developments or products of the Company; and if so requested by the Company, to assign, transfer and convey to the Company all right, title and interest in and to all such inventions and improvements;

Plaintiffs' Exhibit No. 16—(Continued)

(b) At the request and expense of the Company, to make, execute and deliver any and all application papers, assignments or instruments, and to perform or cause to be performed such other lawful acts as the Company may deem desirable or necessary in making or prosecuting applications, domestic or foreign, for patents and reissues and extensions thereof, and to assist and cooperate (without expense to him) with the Company or its representatives in any controversy or legal proceedings relating to said inventions and improvements or the patents which may be procured thereon;

(c) To regard and preserve as confidential all information pertaining to the Company's business or that may be obtained by the Employee from specifications, drawings, blue prints, reproductions and other sources, and not to publish or disclose either during the term of employment or subsequent thereto, without the written approval of the Company, such or any other confidential information obtained by the Employee while in the employment of the Company.

2. The Company, if it considers any invention or improvement reported by the Employee pursuant to paragraph 1 hereof to be of substantial value and patentable, will, after completing its investigation in regard thereto, award and pay to the Employee the sum of Ten Dollars ($10.00).

Plaintiffs' Exhibit No. 16—(Continued)

3. The Company, if it elects to acquire any invention or improvement referred to in paragraph 1 hereof, agrees:

(a) To notify the Employee of its election so to do within nine months from the date of the complete disclosure of such invention or improvement to the Company;

(b) To pay all expenses in connection with the preparation and prosecution of patent applications in the United States of America and all foreign countries wherein the Company may desire to obtain patents;

(c) To pay the Employee an additional cash award of Forty Dollars ($40.00) upon execution by Employee of applications for United States letters patent upon such invention or improvement, together with an assignment thereof to the Company;

(d) To pay to the Employee an additional cash award of Fifty Dollars ($50.00) if and when the Company obtains a United States patent on such invention or improvement, it being understood that no such award will be paid to the Employee in connection with the granting of any foreign patent;

(e) To pay to the Employee for each of the Employee's inventions additional compensation consisting of a percentage of any income derived by the Company from any sale of such invention or part thereof, or from any royalties which the Company may collect from licenses

Plaintiffs' Exhibit No. 16—(Continued)

to others for the use of such invention, on a sliding scale, as follows:

Of the first $1,000 or part thereof... 30%
Of the next $1,000 or part thereof....25%
Of any further sums in excess
of $2,000........................ 20%

4. It is understood and agreed that the obligation of the Company to make payments pursuant to paragraph 3(e) hereof shall continue during the life of any patent subject to this agreement notwithstanding termination of the Employee's employment with the Company, and that in the event of the Employee's decease, such payments will be made to his executors, administrators or representatives.

5. It is further understood and agreed that the Company may report any such invention or improvement to Manufacturers Aircraft Association, Inc., either with or without claim for compensation therefor, or sell such invention or improvement, or license the manufacture thereof for such price or royalty as the Company in its sole judgment and discretion shall determine, or if the Company elects so to do, grant royalty-free licenses for the use of such invention, or waive future royalties for a definite or indefinite period of time on any license theretofore issued by the Company on a royalty basis, and that in any of such events, the Employee shall have no claim or claims against the Company,

Plaintiffs' Exhibit No. 16—(Continued)

except to receive under the provisions of paragraph 3(e) hereof the percentages above set forth of such amounts as the Company shall collect through the sale of such invention or improvement or the issuance of licenses to use the same.

6. If the Company shall fail to elect in writing that it desires to prosecute a patent application on any invention or improvement specified in paragraph 1 hereof within nine months following the complete disclosure thereof to the Company, then all rights of the Company in and to such invention or improvement shall revert to the Employee with the exception only that the Company shall have a free shop right with respect thereto.

7. Neither this agreement nor any benefits hereunder are assignable by the Employee, but the terms and provisions hereof shall inure to the benefit of the Company's successors and assigns.

Dated: September 7, 1942.

CONSOLIDATED AIRCRAFT CORPORATION,
By /s/ H. EUGENE POSEK.

/s/ MAURICE ADOLPH GARBELL,
Employee.

Witness:
/s/ HILDEGARD H. WALTER.
Form 758A (Pat.)

Admitted November 21, 1950.

PLAINTIFFS' EXHIBIT No. 17

Maurice A. Garbell, D. Sc.
Consulting Engineer
1714 Lake Street
San Francisco 21, California
Telephone Bayview 9186

August 5, 1946.

Consolidated Vultee Aircraft Corporation,
San Diego 12, California.

Attention: Mr. Isaac M. Laddon,
Executive Vice-President.

Gentlemen:

It has come to my attention that you have adopted and are utilizing my well-known method of safety wing design in the manufacture of certain commercial and military flying craft.

I am therefore privileged to extend to you at this time an offer to negotiate a license agreement for your use of the aforesaid method of wing design; application for letters patent on the aforesaid invention was filed by me.

I shall look forward to the pleasure of your early reply.

Yours very truly,

/s/ MAURICE A. GARBELL.

MAG:ef

[Stamped]: Received Aug. 8, 1946.

[Attached Envelope]

[27 cents in cancelled U. S. postage stamps.]

[Post-date]: Registered S.F. 8/5/46.

[Post-date]: San Diego 8/7/46.

[Return address]: Dr. Maurice A. Garbell, 1714 Lake Street, San Francisco 21, Calif.

[Addressee]: Consolidated Vultee Aircraft Corporation, San Diego 12, California. Attention: Mr. Isaac M. Laddon, Executive Vice-President.

[Stamped]: Registered No. 45739. Return Receipt Requested.

Admitted November 21, 1950.

Maurice A. Garbell, D. Sc.
Consulting Engineer
1714 Lake Street
San Francisco 21, California
Telephone Bayview 9186

August 12, 1946.

Registered

Consolidated Vultee Aircraft Corporation,
San Diego 12, California.

Attention: Mr. I. M. Laddon, Exec. Vice-Pres.
Mr. G. T. Gerlach, Patent Director.

Gentlemen:

Your letter of August 9th, 1946, is before me.

May I respectfully refer you to my paper "Effective Control of Stalling Characteristics of Highly Tapered and Swept-Back Wings," in the February, 1946, issue of the Journal of the Aeronautical Sciences. This publication states the basic principles underlying my invention concisely, lucidly, and substantially; it also conveys the general scope of my patent application.

I trust that you will find the above-mentioned material helpful in enabling you to evaluate my offer of a license to you.

Yours very truly,

/s/ MAURICE A. GARBELL.

MAG:ef

[Stamped]: Received Aug. 14, 1946.

[Attached Envelope]

[27 cents in cancelled U. S. postage stamps.]

[Post-date]: Registered S.F. 8/12/46.

[Post-date]: San Diego 8/13/46.

[Return address]: Dr. Maurice A. Garbell, 1714 Lake Street, San Francisco 21, Calif.

[Addressee]: Consolidated Vultee Aircraft Corporation, San Diego 12, California. Attention: Mr. Isaac M. Laddon, Executive Vice-President.

[Stamped]: Registered No. 62578. Return Receipt Requested.

PLAINTIFFS' EXHIBIT No. 18

Consolidated Vultee Aircraft Corporation
General Offices
San Diego 12, California

August 9, 1946.

Dr. Maurice A. Garbell,
1714 Lake Street,
San Francisco 21, California.

Dear Sir:

Your letter of August 5th directed to Mr. Laddon has been referred to the writer. Since we are unaware of any method of wing design owned by you and utilized in the design of our airplanes, we are unable to evaluate your offer of a license. If you will let us know in detail the invention you believe we are using, we will be glad to give the matter our prompt consideration.

We will accept a copy of the patent application to which you refer for the purpose of a disclosure, on the basis that in so doing, the disclosure is made to us without obligation based upon any kind of confidential relationship, and that no expressed or implied liability exists except to the extent that the subject matter may later support valid patent claims.

Yours very truly,

CONSOLIDATED VULTEE AIRCRAFT CORPORATION.

/s/ G. T. GERLACH,
Patent Director.

GTG:mm

[Attached Envelope]

[Post-date]: 8/9/46.

[Cancelled U. S. 3 cent stamp.]

[Return Address]: Patent Department, Consolidated Vultee Aircraft Corporation, General Offices, San Diego 12, California.

[Addressee]: Dr. Maurice A. Garbell, 1714 Lake Street, San Francisco, Calif.

Admitted November 21, 1950.

PLAINTIFFS' EXHIBIT No. 19

Consolidated Vultee Aircraft Corporation
General Offices
San Diego 12, California

August 15, 1946.

Dr. Maurice A. Garbell,
1714 Lake Street,
San Francisco 21, California.

Re: Your letter of August 12, 1946 to Mr. I. M. Laddon and Mr. G. T. Gerlach.

Dear Sir:

On December 19, 1944, during your employment at CVAC, you submitted a copy of your paper "Effective Control of Stalling Characteristics of Highly Tapered and Swept-back Wings" to the Patent Department of this company, as a Disclosure

Plaintiffs' Exhibit No. 19—(Continued)

of Invention under the terms of the CVAC Invention Agreement executed by you on September 7, 1942.

Our investigation of this matter indicated (a) that it did not contain material of a patentable nature, and (b) the stall control techniques set forth in the article were well known and amply disclosed in many prior patents. A copy of our Search Report is attached. In view of this, a decision was reached to inactivate the disclosure from the standpoint of CVAC's filing a patent application, and our file indicates that you were verbally informed of this decision.

Under the CVAC Invention Agreement:

Paragraph 6. "If the Company shall fail to elect in writing that it desires to prosecute a patent application on any invention or improvement specified in paragraph 1 thereof within nine months following the complete disclosure thereof to the Company, then all rights of the Company in and to such invention or improvement shall revert to the Employee with the exception only that the Company shall have a paid-up non-exclusive license with respect thereto."

In view of our opinion that no patentability exists in the matter disclosed in your article; and since this company has retained a non-exclusive license to any claim that might be allowed by the Patent Office under the application that you have apparently

Plaintiffs' Exhibit No. 19—(Continued)

filed, there appears no practical purpose in further discussion of our obtaining rights from you. Therefore, unless you believe there is some angle we have overlooked, we will consider the matter concluded.

Yours very truly,

CONSOLIDATED VULTEE
AIRCRAFT CORPORATION,

/s/ G. T. GERLACH,
Patent Director.

GTG:ff
cc: I. M. Laddon
(Copy)

Search Report

June 26, 1946

Re: Docket No. 1562-2,
Airfoil Design Having
Three Controlled Sections,
Maurice A. Garbell.

Report of Search with respect to the above identified disclosure relating to a design means for effecting control of stalling characteristics particularly of highly tapered and swept back wings in which the wing design is based upon the employment of three controlled sections, one located at the wing root, another at the mid-span section, and the third at the wing tip, each section being connected to that next adjacent by straight lines. The desirable results from this design would be a stalling characteristic for the airfoil in which the stall begins initially

Plaintiffs' Exhibit No. 19—(Continued)

at the mid-span section and spreads progressively and evenly inboard and outboard from that point.

The following references appear to present the closest patented are with respect to the present disclosure:

1,246,010	Burgess	11/ 6/17	244-105xr
1,547,644	Cronstedt	7/28/25	244-35
1,729,970	Soldenhoff	10/ 1/29	244-35
1,792,015	Herrick	2/10/31	244-35
1,817,275	Soldenhoff	8/ 4/31	244-35
1,890,079	Focke	12/ 6/32	244-35
2,165,482	Hovgard	7/11/39	244-13
2,281,272	Davis	4/28/42	244-35
2,298,040	Davis	10/ 8/42	244-35
2,329,814	Andrews	9/21/43	244-35
Br. 20,530/09	Vessey	9/ 8/09	B244-35
Br. 573,314	Armstrong-Whitworth	8/28/43	B244-83

The references Armstrong-Whitworth and Burgess each disclose tapered wings having considerable sweep back and which as appears in Fig. 1 of Armstrong and Fig. 4 of Burgess, at approximately the mid-span position have a break in the wing plan form with the outboard portion having at least a different angle of sweep back. The root, mid-span and outboard sections appear to be connected by straight lines but what these sections are or how they might differ from each other is not set forth. The reference showings of Andrews and Davis (Patent No. 2,298,040) are also illustrative of air-foils in which there is a pronounced change, at least in plan form, at approximately the mid-span station.

Plaintiffs' Exhibit No. 19—(Continued)

The references Cronstedt, Soldenhoff, Herrick and Focke have all been noted as providing examples of airfoils in which the root section, mid-span section and outboard section have been specifically set forth and which are of different profile. In Soldenhoff, (Patent No. 1,729,970) the airfoil arrangement of interest is shown in Figs. 2 and 2b while in Herrick, the figures of interest are designated 5 to 8 inclusive. As far as can be determined from the drawings, the various sections would appear to be connected by straight lines. None of these four references sets up the definite object of predetermining a certain desired stalling characteristic of the wing although it might be that one of these designs might have the inherent characteristic of stalling first at the mid-span station.

It is interesting to consider the potentialities of the reference Focke in this respect. In the reference Hovgard, the object is to provide a means for delaying the stall on an inboard section of the wing so that the wing will have a more uniform stall characteristic over all. To obtain this it provides an inboard wing section having one profile and an outboard section having another profile. The Davis Patent No. 2,281,272 may also be of interest as it teaches that a fluid foil may utilize one sectional profile at one point of the span and other section profiles at other points; in the illustration of Fig. 2 there being apparently a basic section located at the mid-span station and root and tip

Plaintiffs' Exhibit No. 19—(Continued)

sections which represent variations from the basic section. Also noted as of possible general interest is the reference Vessey which shows a circular airfoil having a number of different sectional profiles at radial stations about the circumference of the whole unit.

The search has covered the following field:
Class 244, Aeronautics,

Sub-classes 13, Aircraft, heavier-than-air, airplane sustained,
35, Aircraft sustentation, sustaining airfoils,
Br.35, Aircraft sustentation, sustaining airfoils,
45, Aircraft sustentation sustaining airfoils, arrangement.

[Attached Envelope]

Registered

[32 cents cancelled U. S. postage.]

[Post-date]: San Diego 8/16/46.

[Post-date]: S.F. 8/17/46.

[Return address]: G. T. Gerlach, Consolidated Vultee Aircraft Corporation, General Offices, San Diego 12, California.

[Addressee]: Dr. Maurice A. Garbell, 1714 Lake Street, San Francisco 21, California.

[Stamped]: 735809 Registered. Return Receipt Requested. Fee Paid.

Admitted November 21, 1950.

PLAINTIFFS' EXHIBIT No. 21

Consolidated Vultee Aircraft Corporation
San Diego, California

June 17, 1948

Study of Garbell Patent No. 2,441,758 Filed July 16, 1946 and Issued May 18, 1948 Relative to Non-Tip Stall Wing Developed by CVAC

Reference:

(A) Docket No. 1562-2 Method of Airfoil Selection—M. A. Garbell.

(B) Effective Control of Stalling Characteristics of Highly Tapered and Swept-back Wings, by M. A. Garbell C.V.A.C. Dec. 5, 1944. Paper written for presentation before January 1945 Annual Meeting (Cancelled) of the Institute of the Aeronautical Sciences. Received by Patent Dept. December 20, 1944.

(C) Paper of reference (b) corrected slightly and published in the Journal of the Aeronautical Sciences, February 1946.

Summary

1. The only new items or statements in the Garbell patent relative to references (B) and (C) are:

(a) "Additional control sections over three" is obviously design and not invention.

(b) "Greatest mean line camber at wing tip" is a limitation in all claims which is not necessary to the proper functioning of the subject development.

Plaintiffs' Exhibit No. 21—(Continued)

(c) Curvilinear polygon of maximum lift envelopes the spanwise lift distribution is disclosed by the reference (B) and (C) figures but is not named as such.

(d) Errors in the figures of references (B) and (C) have been corrected to some degree in the Garbell patent.

(e) Rough tangent method of locating third control section is only possible addition of "new matter," but it functions only in some circumstances.

2. Claims 1, 2, 3, 5, 6 and 12 appear to be utilized by the Model 240 wing.

The XP5Y-1 does not utilize any of the claims of the Garbell patent.

3. The principle of stall control of tapered plan form wings as disclosed in the Garbell patent is completely shown by the simple addition of plan form taper to the drawings of Cronstedt patent No. 1,547,644 filed in 1921. Claims 11 and 12 of the Garbell patent read on the drawings of the Cronstedt patent.

4. The teaching of the Garbell patent is not followed in the design of the Model 240 wing. The third control section is at 30.7% semi-span outboard of the root section and the stall starts between the fuselage and the engine nacelle at about 15% semi-span.

Plaintiffs' Exhibit No. 21—(Continued)

Pertinent Points of the Development in Ref. (B) Paper

This paper was so incomplete when submitted as a disclosure by Garbell, that the present writer requested that a complete disclosure as required by the "Invention Agreement" be submitted to the Patent Department before it would be accepted for docketing. The paper while based on empirical studies and research that effectively licks the critical wing tip stalling problem of many years standing, does not disclose how to apply the development in good logical technical form as customary with engineering and scientific papers, but rambles on with the faults of conventional wings and what is desired and accomplished with the use of three control sections.

The pertinent points given by the paper follow:

1. Three controlled airfoil sections.

2. The paper does not discuss the relative types of airfoils at the three control sections, except that the "Conclusion" specifies a typical combination of NACA airfoils as follows:

(a) Root Section NACA 2518—2% mean line camber and 18% thickness ratio.

(b) Wing tip or second control section—NACA 4512—4% mean line camber and 12% thickness ratio.

(c) Third control section—NACA 3515—3% mean line camber and 15% thickness ratio. In

Plaintiffs' Exhibit No. 21—(Continued)

this case the second or tip control section has a larger camber than the third control section.

3. Page 7 and figure 9 describe a wing having a wing tip airfoil with a thickness ratio smaller than the optimum for maximum lift so that the optimum thickness airfoil occurs somewhat inboard of the wing tip.

New Statements (Not New Matter) in Garbell Patent Relative to Ref. (B) Paper

1. Additional Control Sections Over Three

   It is an obvious design improvement to use additional control sections if so required by the wing configuration.

2. Greatest Camber at Wing Tip

   Specification column 7, lines 54 to 61 and more specifically lines 59 and 60 "an airfoil section 3 or 8 having the greatest mean-line camber at the tip." Each of the fifteen claims contains this matter as a limitation and the papers (B) and (C) do not discuss the relative cambers of the mean lines of the three control sections.

3. Curvilinear Polygon

   Specification column 7, lines 56 to 61 "defined by the curvilinear polygon 5 (fig. 2), embodying the combination of an airfoil section 1 or 9 having the smallest mean line camber at the tip, and one or more interjacent controlled sections 2 or 11." Claims 2, 3 and 6 contain this matter as a limitation. The "curvilinear polygon" is not

Plaintiffs' Exhibit No. 21—(Continued)

mentioned in reference (B) but figures 7, 9 and 10 disclose it.

4. Figures 2 and 5 Show Stall at Wrong Location
   In figure 7 of reference (B) and (C) papers as drawn, the stall would start at the point of tangency of the two curves near the wing tip. These papers state that the stall starts in mid-semi-span but they do not show how. In figure 2 of the Garbell patent, the stall would occur simultaneously at the two points of tangency of the curves, with the outer stall being localized and the inner stall spreading more rapidly. Figure 3 (ref. specification col. 5, lines 37 to 50) does not agree with figure 2 since it shows the stall starting a little inboard of mid-semi-span. The specification column 10, lines 9 to 25 and more specifically lines 17 to 20, shows how the stall develops at about mid-semi-span and thus corrects the errors in figures 7, 9 and 10 of references (B) and (C) and figures 2 and 5 of the Garbell patent.

5. Specification column 7, lines 14 to 29 and figure 4 disclose a rough method of locating the third control section. This method apparently has no theoretical basis and when applied to figures 7, 9 and 10 of references (B) and (C) erroneously locates the third control section close to the wing tip. Claims 7, 8, 13 and 14 contain this "method" of locating the third control section. The method

Plaintiffs' Exhibit No. 21—(Continued)

fails to work on Model 240 wing since the third control section is at 30.7% semi-span instead of 60 to 80% by this method.

Utilization of Patent Claims by CVAC Models

1. Model 240 Wing
   Root Section NACA 63,4-120 a=1.0
   Mean line camber=.55% Thickness ratio=20%
   Wing tip section NACA 63,4-515 a=1.0
   Mean line camber=2.75% Thickness ratio=15%
   Third control section NACA 63,4-419 a=1.0
   Located at 30.7% semi-span outboard of root section
   Mean line camber=2.2% Thickness ratio=19%
   The mean line camber of the third control section is larger and the thickness ratio is smaller than a straight line fairing between the root and tip sections.
   Claims 1, 2, 3, 5, 6, and 12 appear to be utilized by the Model 240 wing.

2. XP5Y-1 Wing
   Root section NACA 1420
   Mean line camber=1.0% Thickness ratio=20%
   Wing tip section NACA 4412
   Mean line camber=4.0% Thickness ratio=12%
   Third control section NACA 4417 at 60% semi-span
   Mean line camber=4.0% Thickness ratio=17%
   The mean line camber is constant from the third

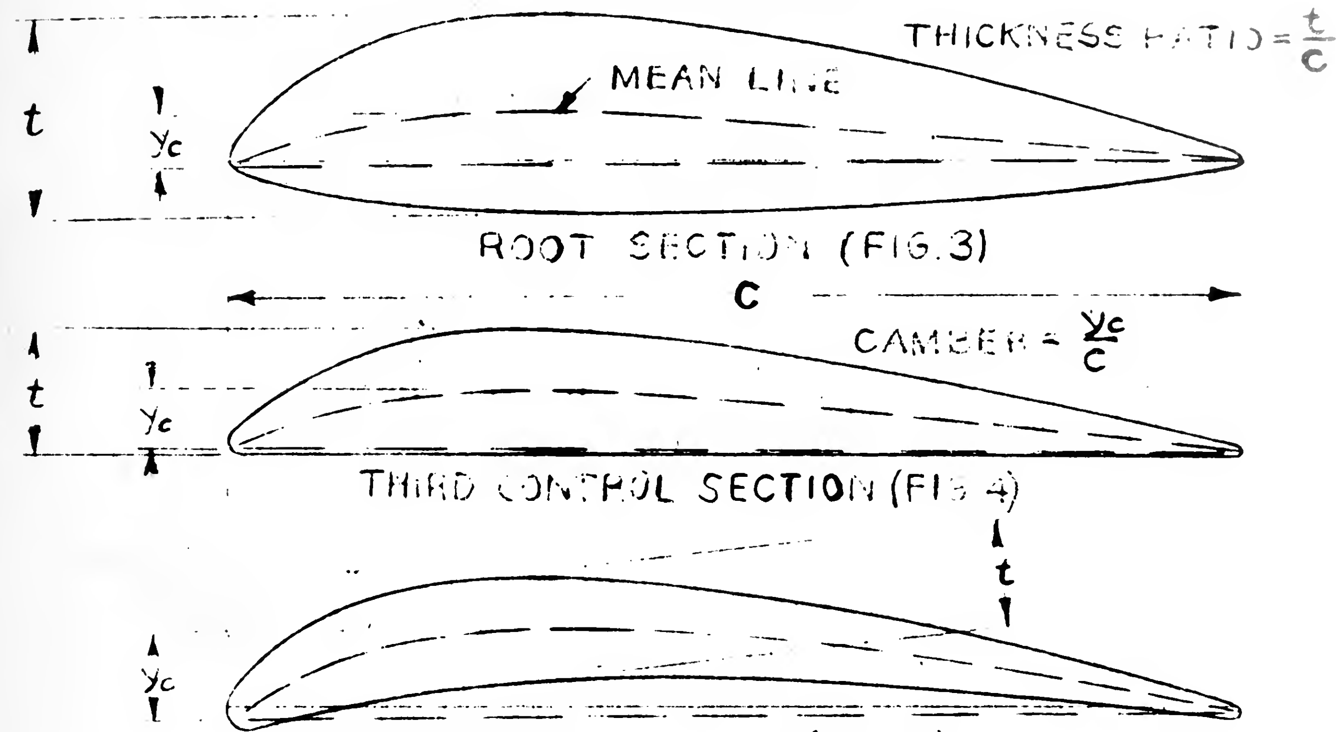
Plaintiffs' Exhibit No. 21—(Continued)

control section to the wing tip. Since each of the fifteen claims contains "the second section with the greatest mean line camber is located at the fluid dynamically effective tip," the XP5Y-1 airplane does not utilize any of the claims of this patent.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION
San Diego, California

CRONSTEDT PATENT NO. 1,547,644 (FILED 1921) DRAWINGS

ANTICIPATE GARBELL PATENT



Figures 3, 4 and 5 of the Cronstedt patent are reproduced above. The mean or median line for each section while not shown in the patent has been developed above. The original chord lines are shown lightly while the mean line chord line is shown heavy (dashed). While the Cronstedt patent specification shows no functioning related to the Garbell patent, figures 3, 4 and 5 clearly show airfoil sections which cause the wing to function closely to the teaching of the Garbell patent. The third control section of figure 4 has a mean line camber greater than that of the root section and less than that of the tip section, and a thickness ratio less than that of the root section and greater than that of the tip section, the same as disclosed by the Garbell patent. Tapering the plan form of the Cronstedt wing is the only change Garbell has made, which certainly is not invention.

Claims 11 and 12 of the Garbell patent read on the wing shown by the drawings of the Cronstedt patent.

Plaintiffs' Exhibit No. 21—(Continued)

Concluding Remarks

1. The teaching of the Garbell patent is not followed in the design of the Model 240 wing. The third control section is located at 30.7% semi-span outboard of the root section and the stall starts between the fuselage and the engine nacelle at about 15% semi-span outboard of the root section.

2. It appears that in some wing designs better stall characteristics can be had by the use of a higher mean line camber for the third control section than for the tip section.

D. A. HALL,

/s/ D. A. HALL.

Admitted November 21, 1950.

PLAINTIFFS' EXHIBIT No. 22

Assignment

Whereas, the undersigned, Maurice A. Garbell, Inc., a corporation organized, existing and doing business under and by virtue of the laws of the State of California, is the owner of an invention relating to certain new and useful improvements in "Fluid Foil Lifting Surface," for which application for Letters Patent of the United States was made on July 16, 1946, Serial No. 683815, and for which said invention Letters Patent of the United

Plaintiffs' Exhibit No. 22—(Continued)

States were duly issued to the undersigned on May 18, 1948, in Patent No. 2,441,758; and

Whereas, the undersigned is likewise the owner of two (2) inventions covering certain new and useful improvements in (1) Fluid Dynamic Stabilizer and Damper, and (2) Lifting Surface, for which applications have been made for Letters Patent of the United States as follows:

(1) Fluid Dynamic Stabilizer and Damper—Serial No. 683814, dated July 16, 1946.

(2) Lifting Surface—Serial No. 697281, dated Sept. 16, 1946.

and which applications are now pending; and

Whereas, Garbell Research Foundation, a general non-profit corporation organized, existing and doing business under and by virtue of the laws of the State of California, having its principal office located in the City and County of San Francisco, State aforesaid, and being formed for the purposes of scientific research for the benefit of mankind, is desirous of acquiring an undivided three-fourths (¾ths) part of the entire right, title and interest in and to said inventions, and each of them, in and through the United States of America, its territories and all countries foreign thereto, and in and to the said Letters Patent, and in and to the said applications for Letters Patent, and in and to any and all Letters Patent of the United States of America and countries foreign thereto which have been or may be granted thereon;

Plaintiffs' Exhibit No. 22—(Continued)

Now Therefore, for and in consideration of the sum of One Dollar ($1.00) and other good and valuable consideration, receipt whereof is hereby acknowledged, the undersigned, Maurice A. Garbell, Inc., a California corporation, by these presents does sell, assign and transfer unto the said Garbell Research Foundation, a corporation, its successors and assigns, the undivided three-fourths (¾ths) part of the entire right, title and interest in and to said inventions, and each of them, in and throughout the United States of America, its territories and all countries foreign thereto, and in and to said Letters Patent No. 2441758, and in and to said application for Letters Patent, Serial No. 683814 and Serial No. 697281, and any and all Letters Patent and extensions thereof of the United States of America and all countries foreign thereto which have been or may be granted on said inventions, or each of them, or any part thereof, or on said applications or any divisional continuing renewal, reissue or other applications based in whole or in part thereon, or based upon said inventions;

To Be Held and Enjoyed by the said Garbell Research Foundation, a corporation, its successors and assigns, for its or their interest, and its or their own use and behoof, and for its or their legal representatives to the full ends of the terms for which said Letters Patent, or any of them, have been granted or may be granted, including the right and any and all rights to commence, maintain and prose-

Plaintiffs' Exhibit No. 22—(Continued)

cute any action and all actions for injunctive or other relief against any infringement thereof, and to recover any profits and/or damages arising out of the infringement of said inventions and/or Letters Patent, or either or any of them, as fully and entirely as the same would have been held and enjoyed by the undersigned had this assignment not been made; and

The undersigned does hereby authorize and request the Commissioner of Patents of the United States of America to issue any and all Letters Patent of the United States of America which may be granted upon the said applications above referred to, or any of them, or upon said inventions or any part thereof to the undersigned and the said Garbell Research Foundation, a corporation, as their interests appear in accordance with the terms hereof; and

The undersigned does agree for itself, its successors and assigns, to execute without further consideration any further or additional legal documents, and any further or additional assignments and any reissue, renewal or other applications for Letters Patent that may be deemed necessary by the assignee herein named fully to secure to the said assignee its interest, as aforesaid, in and to said inventions, or any part thereof, and in and to several Letters Patent, or any of them; and

The undersigned does hereby covenant for itself and its legal representatives, and does hereby agree

Plaintiffs' Exhibit No. 22—(Continued)

with the said Garbell Research Foundation, a corporation, its successors and assigns, that the undersigned has granted no license to make, use or sell the said inventions, or either of them, or any part thereof; that prior to the execution of this assignment, its right, title and interest in said inventions, and each of them, had not been encumbered; that it then had and does now have good right and title to the same, and that it has not executed nor will it hereafter execute any instrument in conflict therewith.

In Witness Whereof, the undersigned has hereunto set its hand and seal this 15th day of September, 1949.

[Seal] MAURICE A. GARBELL, INC.

By /s/ ETTA FEITELBERG,
Vice-President and Treasurer.

/s/ LOLA J. FEITELBERG,
Secretary.

State of California,
City and County of San Francisco—ss.

On the 15th day of September, 1949, before me, Theodore A. Kolb, Notary Public in and for the City and County of San Francisco, State of California, personally appeared Etta Feitelberg and Lola J. Feitelberg, known to me to be the Vice-President-Treasurer and Secretary respectively of

Plaintiffs' Exhibit No. 22—(Continued)

the corporation that executed the within instrument, and acknowledged to me that such corporation executed the same.

[Seal] /s/ THEODORE A. KOLB,
Notary Public in and for
Said County and State.

My Commission Expires January 5, 1953.

Recorded, U. S. Patent Office Sept. 20, 1949. Liber N221, Page 123.

Assignment

Whereas, I, Maurice A. Garbell, of the City and County of San Francisco, State of California, have invented certain new and useful improvements in (1) Fluid Dynamic Stabilizer and Damper, (2) Fluid Foil Lifting Surface, and (3) Lifting Surface, for which I have made application for Letters Patent of the United States as follows:

(1) Fluid Dynamic Stabilizer and Damper—Serial No. 683,814, dated July 16, 1946

(2) Fluid Foil Lifting Surface—Serial No. 683,815, dated July 16, 1946

(3) Lifting Surface—Serial No. 697,281, dated Sept. 16, 1946

and which applications are now pending; and

Whereas, Maurice A. Garbell, Inc., a California corporation, with its principal place of business in

Plaintiffs' Exhibit No. 22—(Continued)

the City and County of San Francisco, State of California, is desirous of acquiring the entire right, title and interest in and to said inventions in and throughout the United States of America, its territories, and all countries foreign thereto, and in and to the said applications for Letters Patent, and in and to any and all Letters Patent of the United States of America and countries foreign thereto, which have been or may be granted thereon:

Now, Therefore, for and in consideration of the sum of One Dollar ($1.00) and other good and valuable consideration, receipt whereof is hereby acknowledged, I, Maurice A. Garbell, do hereby sell, assign and transfer unto the said Maurice A. Garbell, Inc., its successors and assigns, the entire right, title and interest in and throughout the United States of America, its territories, and all countries foreign thereto, in and to said inventions, said applications for Letters Patent, Serial No. 683,814, 683,815 and 697,281, respectively, and any and all Letters Patent and extensions thereof, of the United States of America and countries foreign thereto, which have been or may be granted on said inventions or any part thereof, or on said applications or any divisional, continuing, renewal, reissue, or other applications based in whole or in part thereon, or based upon said inventions:

To be held and enjoyed by the said Maurice A. Garbell, Inc., its successors and assigns, for its or their interest, for its or their own use and behoof,

Plaintiffs' Exhibit No. 22—(Continued)

and for its or their legal representatives, to the full ends of the terms for which said Letters Patent, or any of them, have been granted or may be granted, as fully and entirely as the same would have been held and enjoyed by me had this assignment and sale not been made; and

I do hereby authorize and request the Commissioner of Patents of the United States of America to issue any and all Letters Patent of the United States of America which may be granted upon the said applications above referred to, or any of them, or upon said inventions or any part thereof, to the said Maurice A. Garbell, Inc.; and

I do hereby agree, for myself and for my heirs, executors and administrators, to execute without further consideration, any further legal documents and any further assignments, and any reissue, renewal or other applications for Letters Patent that may be deemed necessary by the assignee herein named, fully to secure to the said assignee its interest as aforesaid in and to said inventions or any part thereof, and in and to several Letters Patent, or any of them.

And I do hereby covenant for myself and my legal representatives and agree with Maurice A. Garbell, Inc., its successors and assigns, that I have granted no license to make, use or sell the said inventions, that prior to the execution of this deed my right, title and interest in said inventions had not been encumbered, that I then had good right and

Plaintiffs' Exhibit No. 22—(Continued)

title to the same, and that I have not executed and will not execute any instrument in conflict therewith.

In Witness Whereof, I have hereunto set my hand and seal this 16th day of April, 1948.

/s/ MAURICE A. GARBELL.

State of California,
City and County of San Francisco—ss.

On this 16th day of April, 1948, before me personally appeared Maurice A. Garbell, to me known, and known to me to be the person described in and who executed the foregoing Assignment, and he duly acknowledged to me that he executed the same for the use and purposes therein mentioned.

/s/ VIOLET NEUENBURG,
Notary Public.

Notary Public in and for the City and County of San Francisco, State of California.

My Commission expires January 3, 1951.

Recorded U. S. Patent Office April 20, 1948. Liber S215, Page 545.

Admitted November 22, 1950.

PLAINTIFFS' EXHIBIT No. 25

Intra-Company Correspondence

Consolidated Vultee Aircraft Corporation
General Offices: San Diego, California

Aero Memo #604
Date: March 2, 1945

To: Mr. T. P. Hall

From: Mr. M. A. Garbell

Subject: Alternate Wing for the Model 37 Airplane

Reference: (a) Report entitled: "A Study of Various Alternate Designs to Improve the Stalling Characteristics of the Model 37 Airplane."

Enclosure: (A) Report of reference (a).

The enclosed report presents the results of a study that is intended to correct the now unfavorable stalling characteristics of the XB-36 wing. The object of the study is the attainment of good stalling characteristics, with full lateral control through the stall and adequate stall warning, but at no additional drag penalty over the present XB-36 wing.

The study was undertaken in anticipation of the increasingly stringent stability and control requirements for the commercial 320,000-lb. version of the Model 37 airplane, and in view of the structural redesign required for the recently increased gross weight of that airplane.

The "tri-section wing" principle which has been successfully applied to the Tailless design, the executive transport, and the XB-46 design, yields several satisfactory wings.

None of the proposals requires a change in plan form nor in wing-root thickness, but the airfoils have been altered considerably. The following synopsis correlates the present wing and the two most promising proposals:

	Station			
Wing	Root	60% Span	Tip	
Original XB-36 Wing	63,4-422	63,4-(.43)20.6	63,4-517	Airfoil Section
	Basis	0.25°	0.81°	Aerodynamic Washout
Proposal #6	63,4-222	65,3-518	65,3-514	Airfoil Section
(preferred)	Basis	0.42°	0.42°	Aerodynamic Washout
Proposal #2	63,4-222	63,4-518	63,4-514	Airfoil Section
(2nd choice)	Basis	0.49°	0.49°	Aerodynamic Washout

No attempt has been made in the enclosed report to evaluate, in the light of the CAB requirements on proper stall characteristics, the advantages gained by eliminating a vicious wing-tip stall and increasing the maximum wing lift coefficient by approximately 0.1, because these advantages are self-evident.

It is suggested that an alternate wing be built for the 1/26-scale wind tunnel model of the Model 37 airplane. This model should be tested in one of our forthcoming Galcit or M.I.T. test periods, whenever the opportunity for one day's testing arises. The brief test will provide preliminary information on the improved alternate wing, should further wind-tunnel and flight tests confirm the unfavorable stall characteristics of the XB-36 wing.[1]

/s/ M. A. GARBELL.

MAG:jm
cc: Dev. Engr. File

[[1]Longhand note referring to this paragraph]: Not at this time. /s/ P. T. H.

Admitted November 22, 1950.

Consolidated Vultee Aircraft Corporation
San Diego, California

Date: March 3, 1945.

Memo
R. L. Bayless

To: Mr. T. P. Hall.

This study was done over a period of time as other work permitted.

The proposed changes are based on airfoil data and theoretical analyses which were not available at the time the XB-36 wing was established.

R. L. BAYLESS.
/s/ B.

[Marginal note]: Miss C pl bring up on my return plus 2 days.

/s/ T. P. H.

PLAINTIFFS' EXHIBIT No. 26

Intra-Company Correspondence
Consolidated Vultee Aircraft Corporation
General Offices . . San Diego, California

Date: April 24, 1945.

To: M. A. Garbell, Development Engineering, San Diego.
From: Patent Department.
Subject: Docket 1128-R
Hydrofoil
Docket 1129-R
High Speed Air Intake.

Dear Mr. Garbell:

We have been notified by our Accounting Department that two checks each in the sum of ten dollars ($10) have been forwarded to you.

This is in accordance with paragraphs 2 and 3(a) of the CVAC Invention Agreement and is notification to you of the election of the company to accept the inventions involved.

/s/ WALTER J. JASON.

WJJ:jn

Plaintiffs' Exhibit No. 26—(Continued)

Intra-Company Correspondence
Consolidated Vultee Aircraft Corporation
General Offices . . San Diego, California

Date: November 20, 1944.

To: M. A. Garbell—661-379745—San Diego Development Engineering.

From: Patent Department.

Subject: Docket 1129-P
High Speed Air Intake
M. A. Garbell.

We have received your disclosure on the High Speed Air Intake which has been assigned Docket No. 1129-P. You will be advised of the results of our investigation as soon as possible.

If you should have any further inquiries, suggestions or additions, please contact Mr. Rolf Evers. Division Patent Engineer.

/s/ GORDON GRENOLDS,
Patent Department.

GG/abh
cc: J. L. Kelley
R. Evers

Plaintiffs' Exhibit No. 26—(Continued)

Intra-Company Correspondence
Consolidated Vultee Aircraft Corporation
General Offices .. San Diego, California

Date: December 18, 1944.

To: Mr. Rolf Evers, Patent Engineer.
From: Mr. M. A. Garbell.
Subject: Docket 1129-P—High-speed Air Intake.
Reference: (a) Mr. Walter J. Jason's Memo of December 9, 1944.

A study of the patents enclosed with the referenced memo has been completed and the following conclusions have been reached:

Wagenseil 1,376,178

This patent refers to the now conventional air intake. The undesirable airflow characteristics of this and similar intakes has been already discussed in the subject disclosure. The bodies located in the intake and exhaust of the duct, respectively, shown in Fig. 6 of Mr. Wagenseil's patent application, are merely control organs (valves). The valves are evidently not intended to provide any favorable pressure distribution for a more efficient air inflow, free from airflow separation at moderate and large angles of attack.

Newcombe 2,353,966

The small airfoil shaped body located in the leading-edge duct of radiator 25 in Fig. 1 of Mr. Newcombe's patent application does not have an appropriate shape to prevent separation at the duct lips. It is totally contained in the basic airfoil shape,

Plaintiffs' Exhibit No. 26—(Continued)

where as the subject invention utilizes the aerodynamic pressure distribution and circulation around a protruding airfoil to convey the air more efficiently into the air intake duct. Mr. Newcombe's patent does not contain any claim regarding the aerodynamic action of such an airfoil.

Townend 1,813,645

This patent describes an annular cowling about a circular body from which individual cylinders are protruding into the airstream. The device, known as the "Townend ring" provides an external fairing of an aerodynamically rough body, rather than a guiding vane of an aerodynamically critical duct.

Vance 2,136,403

This arrangement, intended to achieve a large angle-of-attack range for air intake ducts, attains its goal at a substantial sacrifice in efficiency, because only one of the two branch ducts is fully effective at any large angle of attack (see lines 32 to 35 in the right-hand column of page 3 of the claim).

Dornier 2,249,984

The guide plate 4, shown in this claim, is mainly intended to provide a fairly efficient intake shape when the duct is only half extended. The guide plate is not properly shaped to produce the aerodynamic pressure distribution and circulation required to prevent airflow separation aft of the intake lips at moderate and large angles of attack. The guide plate is fully contained in the duct.

Plaintiffs' Exhibit No. 26—(Continued)

Griswold 2,348,253

The thermal exchange elements 111 (Figs. 9, 10, 11, and 11a), 227 (Fig. 20), and 248 (Fig. 21), fully contained in the duct and located well aft of the air intake, exert no aerodynamic action other than the thermodynamic transfer of energy from the radiator into the airstream.

Conclusion

It is apparent that none of the patents enclosed with the reference memo evidences any of the important aerodynamic features of the subject disclosure. The guide vanes or plates shown in some of these patents are not aerodynamically integral parts of the intake ducts.

No claim is contained in any of the aforementioned inventions that a high-speed air intake may include a properly designed leading-edge airfoil to prevent intake-lip separation while retaining full duct efficiency throughout an ample range of angles of attack.

It is suggested that the principle employed in the subject invention is of sufficient generality to warrant ample patent protection.

The air-intake design proposed in the subject disclosure will be tested in a forthcoming wind-tunnel test. It is believed that patent protection should be secured prior to the wind-tunnel test in order to avoid difficulties which may arise as a consequence of manipulation of the duct by other than CVAC personnel.

M. A. GARBELL.

MAG:ph

Plaintiffs' Exhibit No. 26—(Continued)

High Speed Air Intake

M. A. Garbell (661-379745)—Inventor

This invention relates to aircraft and particularly to aircraft having air intake openings or scoops in the leading edges of their wings or nacelles.

Airplanes are designed with air intake openings in the leading edges of their wings, and it has been found that when such airplanes are flown in their normal angle of attack the air will be effectively rammed directly into the intake openings. When the angle of attack is increased, however, the air instead of being forced directly into the opening with uniform pressure will flow across the lower edge of the opening at an angle thereto. For example, when the airplane is flown with a normal angle of attack the air will be rammed into the intake opening or scoop in the direction indicated by the arrow in Figure 1 of the drawings. When the attack angle is increased as shown in Figure 2, the air will enter the intake opening in the direction indicated by the arrow in this view. When this latter condition exists the air will tend to separate within the opening at the rear of its lower lip, causing turbulent flow which reduces the ram recovery and hence the pressure drop available for optimum volumetric flow of the air used for oil cooler and intercooler intakes and other purposes.

To overcome this condition the inventor has devised means, in the form of an aerodynamic body arranged within the opening or scoop, and adapted to direct the flow of air into the opening in such a

Plaintiffs' Exhibit No. 26—(Continued)

way that separation or turbulence is prevented, a more appropriate pressure distribution obtained to achieve efficient diffusion, and the required flow of air through the ducts produced with the least loss in ram efficiency.

Figure 3 shows the leading edge of an airplane wing 2 having an elongate opening or scoop 3 for receiving air which is transmitted through a duct to the carburetor or supercharger. In accordance with this invention, the opening 3 is made somewhat wider than ordinary air scoops and extending across the opening from end to end is an intake vane 5. Figure 4 is a cross section on line 4—4 of Figure 3 and shows the intake vane 5 as of aerodynamic shape capable of producing favorable pressure distribution. The aerodynamic body or intake vane 5 is used for the purpose of directing the flow of air into the scoop 3 in such a way that separation or turbulence is avoided. As shown in Figure 3, when the airplane is flown at its normal angle of attack the air will flow across the aerodynamic surface of the body 5 as shown by the arrows to effectively distribute the pressure and properly supply the air ducts. When the angle of attack is increased as shown in Figure 5 the air will be rammed into the scoop 3 in the manner indicated by the arrows in this view. Through the arrangement and shape of the opening and aerodynamic body the pressure of the air passing into the air duct is properly distributed, and the air will thus flow at a high velocity without the occurrence of separation or turbulence adjacent the edges or lips of the air

Plaintiffs' Exhibit No. 26—(Continued)

scoop. While the opening 3 is shown as substantially rectangular in outline, it will be understood that it may assume other shapes, and in this case the aerodynamic body would be of corresponding cross sectional shape.

Date: November 17, 1944.

/s/ MAURICE A. GARBELL,
Inventor.

Date: November 17, 1944.

/s/ W. J. STEVENSON,
Witness.

PLAINTIFFS' EXHIBIT No. 27

Intra-Company Correspondence
Consolidated Vultee Aircraft Corporation
General Offices . . San Diego, California

Date: November 20, 1944.

To: M. A. Garbell—661-379745—San Diego
Development Engineering.

From: Patent Department

Subject: Docket 1128-R
Hydrofoil

Reference: M. A. Garbell.

We have received your disclosure on the Hydrofoil which has been assigned Docket No. 1128-R. You will be advised of the results of our investigation as soon as possible.

If you should have any further inquiries, suggestions or additions, please contact Mr. Rolf Evers, Division Patent Engineer.

/s/ GORDON GRENOLDS,
Patent Department.

GG/abh
cc: R. Evers
J. L. Kelley

Plaintiffs' Exhibit No. 27—(Continued)

Consolidated Vultee Aircraft Corporation
General Offices . . San Diego, California

19 December, 1944.

Mr. Rolf Evers, Patent Engineer.
Mr. M. A. Garbell.
Docket 1128-R—Hydrofoil Arrangement.
Mr. Walter J. Jason's Memo of December 14, 1944.

The patents enclosed with the referenced memo have been studied and the following conclusions have been reached:

Diehl 2,255,046

The writer is in substantial agreement with Mr. Jason's statement, with one important exception. The subject invention relates to an airplane in which the hydrofoils contribute little, if any, static buoyancy. By far the greatest part of the static buoyancy is contributed by the fuselage-hull. Mr. Diehl's invention by contrast, refers to buoyant floats.

Brush	2,073,864
Dyer	1,108,891
Kemp	1,728,937
and others.	

These patents propose merely the use of hydrofoils for hydrodynamic lift instead of floats and hulls for static buoyancy. Hydrofoil arrangements of the types proposed in these patents are unsatisfactory, because the hydrofoils are unable to "break" through the water surface owing to cavitation.

Plaintiffs' Exhibit No. 27—(Continued)

It is the express purpose of the subject invention to overcome this serious deficiency of the older hydrofoil arrangements by means of the high trim angle of the main fuselage-hull.

The high angle of attack shown in Fig. 1 of Dyer's patent claim is not the trim angle of the hull, but merely serves to illustrate a typical take-off attitude of the craft.

Parker 2,347,841

This invention refers to retractable hull steps (not spoilers). There is no direct relation between the subject disclosure and Parker's patent. Such retractable steps have not evidenced the drag reduction anticipated by their inventor.

Additional Remarks on the Subject Disclosure

It is contended that the subject disclosure covers a patentable field of considerable amplitude.

No immediate laboratory tests are contemplated, nor are they believed to be required to demonstrate the patentability of the fundamental principle covered by the disclosure. Airplane designs varying in many secondary features may be developed to accomplish the fundamental intent of the disclosure. It may also be argued that it may not be opportune, in the interest of complete protection for the Company, to have designs employing the principle of the subject invention tested in Government or University owned Research Laboratories prior to filing a patent. It is therefore recommended that patent protection commensurate with the manifest merit of

Plaintiffs' Exhibit No. 27—(Continued)

the subject disclosure be secured before tests of any specific design are initiated.

M. A. GARBELL.

MAG:lm.

Hydrofoil Arrangement for Airplanes

M. A. Garbell (661-379745)—Inventor

This invention relates to aircraft and particularly to an improved seaplane having substantially the same aerodynamic characteristics as a land plane. Specifically, this invention relates to an improved hydrofoil installation on fuselages with high trim angles capable of overcoming the critical sub-surface cavitation period which, heretofore, has presented a serious obstacle to the emergence of hydrofoils.

One object of this invention is to provide an aircraft of conventional type with hydrofoils of appropriate contour mounted rigidly or retractably on the fuselage and/or wings in the approximate location of the ordinary tricycle landing gear.

Another object is to provide an aircraft of this type which is designed to trim at a high angle of trim when taxiing on the surface of the water, and also having one or more spoilers attached to the bottom of the fuselage afterbody.

Another object is to provide an aircraft of this type in which the hydrofoils are adapted to emerge from the water due to the high trim angle of the fuselage, this movement being partly or totally independent of the dynamic lift of the hydrofoils.

Another object is to provide an aircraft in which the spoilers, arranged in the lower portion of the

Plaintiffs' Exhibit No. 27—(Continued)

fuselage afterbody, permit the ship to plane (float dynamically) on two main hydrofoils and auxiliary hydrofoil.

Another object is to provide an aircraft which is adapted to land on the hydrofoils and thereafter settle on its fuselage which forms the hull of the ship.

In the accompanying drawings:

Figure 1 shows a conventional type airplane equipped with hydrofoils and spoilers with the ship shown floating on its fuselage-hull;

Figure 2 shows the ship taxiing at the high trim angle of the fuselage-hull and the spoilers operated to break up the suction between the hull and water in order to permit the transition to hydrodynamic planing on the hydrofoils at a reduced trim angle; and

Figure 3 shows the ship planing on the main and auxiliary hydrofoils just prior to take-off of the ship from the water.

The seaplane herein shown comprises a fuselage 2 of a shape similar to those of conventional airplanes and constituting the hull. The ship may have high wings 3 and engines 4 mounted on the wings to position the propellers 5 (or other propulsion devices) high above the free water surface. Projecting downwardly from the nose of the fuselage 2 in the approximate location of the usual nosewheel is an auxiliary hydrofoil 7 which, as shown in the drawings, is of appropriate shape to produce

Plaintiffs' Exhibit No. 27—(Continued)

high lift and low drag. This hydrofoil may be supported in suitable manner. Below the wings 3 are main hydrofoils 8, also of appropriate contour. These hydrofoils may be suspended from the sides of the fuselage-hull 2 as shown or from the under surface of the wings 3. Arranged in the aft portion of the fuselage are retractable spoilers 10.

Assuming that the fuselage-hull 2 of the ship is floating on the surface of the water as shown in Figure 1 with the engines running, when it is desired to take off from the water the thrust of the propulsion devices 5 is increased, and the ship will move forwardly. Because of the high fuselage trim angle, as the ship gains speed, the hydrofoils will climb upwardly with the auxiliary hydrofoil 7 finally emerging from the water and the main hydrofoils 8 planing along the surface of the water as shown in Figure 2. At this juncture the aft portion of the fuselage-hull 2 will be in contact with the surface of the water and to reduce the fuselage trim angle the spoilers 10 are lowered as shown in Figure 2. The spoilers 10 form, in effect, a step similar to that usually provided in the bottom of conventional seaplane hulls so that the suction between the aft section of the fuselage and the water is quickly overcome and the ship thus permitted to plane or float dynamically on the main and auxiliary hydrofoils as shown in Figure 3. The seaplane is thus free to plane on the surface of the water, and as its forward motion is increased to the necessary degree, the craft will take off from the water.

Plaintiffs' Exhibit No. 27—(Continued)

Because of the shape of the hydrofoils the resistance or drag imposed thereby will be reduced to a minimum, and the craft may be flown in a manner similar to conventional airplanes.

When it is desired to land on the water the ship is brought down in such a manner that the hydrofoils 7 and 8 will plane along the surface of the water, and by gradually reducing the speed of the ship and trimming the elevators and other control surfaces the hydrofoils will submerge and the fuselage-hull 2 finally settle on the surface of the water as shown in Figure 1.

It will be observed from the foregoing that the present invention provides a seaplane having a hull (fuselage) and planing surfaces (hydrofoils) of desirable aerodynamic and hydrodynamic shapes, thereby avoiding the use of large, heavy and drag-producing hulls now used in seaplanes. In addition, the high fuselage trim angle acts to overcome the critical sub-surface cavitation of the hydrofoils, the spoilers permitting the airplane subsequently to continue planing on the hydrofoils, and the ship will be able to take off from the water with minimum travel.

Date: November 17, 1944.

/s/ MAURICE A. GARBELL,
Inventor.

Date: November 17, 1944.

/s/ W. J. STEVENSON,
Witness.

PLAINTIFF'S EXHIBIT NO. 28

Admitted November 22, 1950.

Dec. 27, 1949 H. A. SUTTON ET AL 2,492,245

AIRCRAFT CONTROL MEANS

Filed July 25, 1945 2 Sheets-Sheet 1

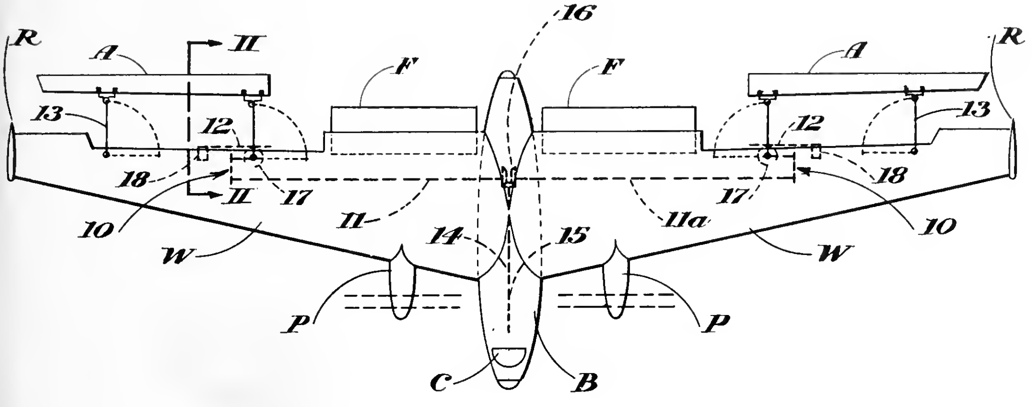


Fig. 1

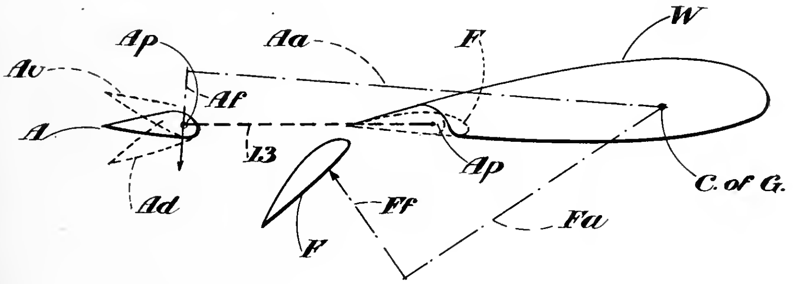


Fig. 2

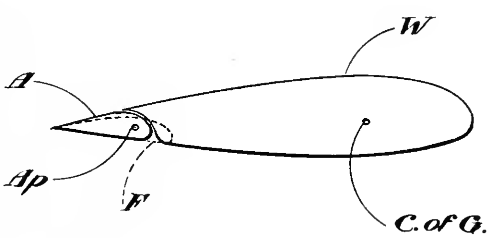


Fig. 3

H. A. Sutton INVENTOR.
and Rolf Evers
BY James M. Clark
Their Patent Attorney

Dec. 27, 1949 H. A. SUTTON ET AL 2,492,245

AIRCRAFT CONTROL MEANS

Filed July 25, 1945 2 Sheets-Sheet 2

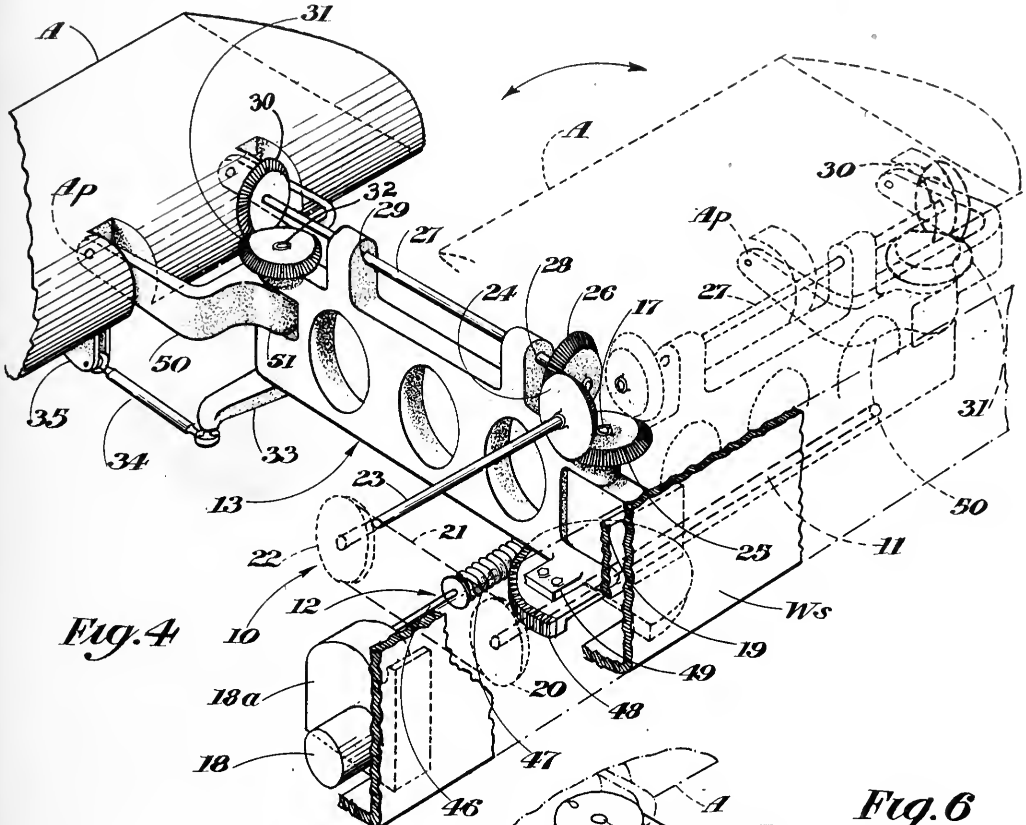


Fig. 4

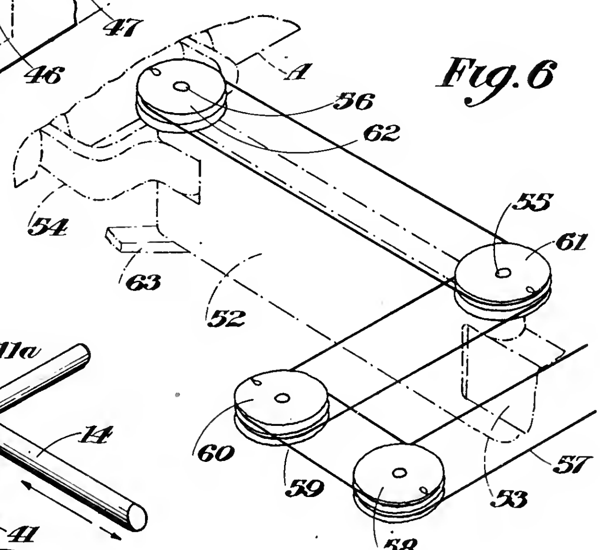


Fig. 6

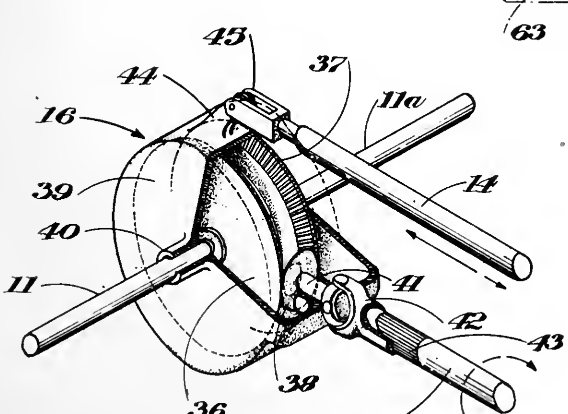


Fig. 5

H. A. Sutton INVENTORS
and Rolf Evers
BY James M. Clark
Their Patent Attorney

ented Dec. 27, 1949

2,492,245

UNITED STATES PATENT OFFICE

2,492,245

AIRCRAFT CONTROL MEANS

Harry A. Sutton, Baltimore, Md., and Rolf Evers, Coronado, Calif., assignors to Consolidated Vultee Aircraft Corporation, a corporation of Delaware

Application July 25, 1945, Serial No. 606,914

3 Claims. (Cl. 244—13)

1

present invention relates to the control craft and like vehicles, and is more par-ly directed to extensible control surfaces ed for both longitudinal and lateral control. use of high lift flaps for take-off and land-rposes has produced decided aerodynamic tages, particularly in the operation of large ft. The use of certain types of such flaps, er, presents a number of problems par-ly from their inherent disadvantage of g a relatively large rearward shift in the of lift of the wing when the flap is ex-l rearwardly. In aircraft of the conven-type having rearwardly disposed horizontal rfaces, this disturbance in the location of enter of lift is readily accommodated by cing a negative lift or downward force by rizontal tail surfaces. In the conventional nage type airplane the flaps usually pro-down-wash effect which acting upon the zer tends to counterbalance the diving nt and thereby produce a stable condition. egative lift in the tail surfaces, however, o the load, or to the lift to be developed main sustaining surface, and to this ex-has been found objectionable and to de-rom the aerodynamic efficiency and load-aracteristics of the airplane.

mpts have been made in the prior art to me these disadvantages by the provision xiliary lifting surfaces located forwardly espect to the center of gravity of the air-or the center of pressure of the wing in that this auxiliary lift assist the lift of the sustaining surface, rather than to add un-o its load.

ail-less, or flying wing, types of aircraft the certain type flaps have presented problems are not readily solved as by taking advan-f the use of a conventional tail surface, and uselage projection forward of the wing's g edge in tail-less models is not always s to support a forwardly disposed auxiliary surface. Several efforts have been made l-less type airplanes to provide suitable for balancing the diving moments created use of these high lift flaps, but such prior have either been relatively unsuccessful, esulted in materially complicating the de-f the control system or have been found onable for other reasons.

present invention relates to an improved l means for providing a balancing force to ract the diving moment produced in air-provided with flaps and is particularly

2

adapted to the balancing of these diving moments and the provision of longitudinal control and stability in tail-less or flying-wing types of aircraft. The improved surface comprising the present invention consists essentially of a rearwardly and outwardly extensible control surface which is operable in both its retracted and extended, as well as all of its intermediate, positions—both differentially or simultaneously opposite for use as an aileron in providing lateral control, and simultaneously in the same direction, either upwardly or downwardly, for use as an elevator to obtain longitudinal control. The invention further consists in novel actuating mechanism by means of which the control surface is extended from its position at the trailing edge of the main sustaining surface and by which it is concurrently or differentially controlled at the will of the pilot.

It is accordingly a major object of the present invention to provide a control surface which is extensible from its normal position at the trailing edge of the wing, both rearwardly and outwardly away from the longitudinal plane of symmetry of the aircraft. It is a further object to provide such an extensible control surface which is particularly adapted for use with airplanes of the tail-less or flying wing type and in which the surface is controllable in both its normal retracted and extended positions. It is a still further object to provide mechanism for the concurrent extension of a pair of such control surfaces which mechanism is such that these surfaces may be supported for their operation in any position intermediate their retracted and extended positions.

It is also an object of this invention to provide an extensible surface which is capable of use as an aileron for lateral control and as an elevator for longitudinal control. It is a further object to provide actuating mechanism for the differential operation of said surface as an aileron in each of its extended positions and for its simultaneous operation as an elevator. A further object resides in the provision of such a combined aileron-elevator surface which is appreciably extended outwardly from the plane of symmetry of the aircraft to improve its effectiveness as an aileron and which is extended rearwardly from the center of gravity of the airplane to increase the effectiveness of the surface as an elevator. Other objects and advantages of this invention will become apparent to those skilled in the art after reading the present specification together with the drawings forming a part hereof in which:

Fig. 1 is a plan view of a tail-less type airplane to which the present invention has been applied;

Fig. 2 is a transverse sectional view taken through the wing, the flap and the extended balancing surface along the line II—II of Fig. 1, showing diagrammatically the nature of the forces developed by the flap and the improved balancing surface;

Fig. 3 is a transverse sectional view, similar to Fig. 2, showing the flap and balance surface in their retracted positions;

Fig. 4 is an isometric view of the operating mechanism by which the auxiliary surface is extended and rotated into its operating positions;

Fig. 5 is a similar isometric view of a form of control mechanism by which the pair of auxiliary surfaces is differentially or concurrently actuated; and

Fig. 6 is an isometric view of a modified form of the mechanism shown in Fig. 4 but in which the rotation of the auxiliary surface is cable controlled.

Referring now to Fig. 1, there is shown a plan view of an airplane of the tail-less type provided with a body or fuselage B, having a control cabin or cockpit C and a main sustaining surface or wing W. While the present invention has been shown and described as particularly adapted for use with tail-less or flying wing types of aircraft, it is pointed out that this invention is not limited to use therewith. The airplane may preferably be provided with power plants P driving tractor propellers, as well as vertical surfaces R at the wing tips for rudder or steeering control, and high lift flaps F for landing and take-off purposes. It will also be understood that the flaps F may extend fully beneath the fuselage as a continuous auxiliary lift member, or the airplane may be of the flying wing type in which there is no fuselage as such, and the pilot control position may be housed entirely within the wing.

The improved control surface of the present invention is indicated in Fig. 1 by the letter A as shown in full lines in its rearwardly and outwardly extended position. Its operating mechanism is indicated generally in dotted lines by the numeral **10**, with the mechanism for rocking the surface indicated at **11**, and the mechanism for extending the surface indicated as at **12**. Both the flap F and the balancing surface A are shown in their rearwardly extended positions in Fig. 1, as well as by the full line portions of these surfaces in the cross-sectional view in Fig. 2. In the latter figure the C. of G., or center of gravity, is indicated with respect to the wing profile W and the broken line F*a* extending downwardly and rearwardly therefrom indicates the moment arm of the positive lifting force F*f* developed by the extended flap F. Similarly, the rearwardly extending line A*a* from the C. of G. toward the balancing surface A represents the moment arm of the negative force A*f* developed by the balancing surface.

The flap F is projectable in a well known manner rearwardly and downwardly from the dotted line position in which it is nested within an undersurface recess in the wing W to the extended position shown in full lines in Fig. 2. The balancing surface A is extendable upon a bracket assembly indicated generally by the dotted line **13**; in which extended position it is rockable about the axis of its pivot A*p* into the upper dotted position A*u*, and through its neutral or normal position into its lower or downward position A*d*. It will accordingly be noted that as the flap F is extended for take-off, landing or other flight condition it [de]velops a positive lifting force or moment abou[t the] center of gravity of the aircraft (C. of G.) [to] cause the same to dive, but that this force ca[n be] balanced by a relatively smaller downwar[d] negative lifting force acting through a longer [mo]ment arm developed by the control of the bal[anc]ing surface A. In the neutral retracted positi[on] of the flap F and the auxiliary balance surfac[e] they both form the trailing portions of the [wing] W as shown in Fig. 3.

As indicated generally in Fig. 1 the mecha[nism] for rocking the auxiliary balance surface A a[bout] its pivot comprises the push-pull and to[rque] shafts **14** and **15**, respectively, which extend [rear]wardly through the fuselage B from the pilot [po]sition at C to a conversion unit **16** from w[hich] torque shafts **11** extend laterally spanwise o[f the] wing to the actuating mechanism generally [lo]cated at **10** in the region of the vertical piv[ot] of the surface supporting bracket **13**. The [bal]ance surfaces A are preferably projected [to] their extended positions by means of a mo[tor] controllable from the pilot position at C and [oper]ating the swinging brackets **13** through the a[bove] mentioned mechanism indicated generally [at] and to be further described in detail below in [con]nection with Fig. 4.

Referring now to Fig. 4, a rear spar or [other] spanwise structural element of the wing is [lo]cated at W*s* and has fixedly attached to the [rear] side thereof a pair of brackets **19** which are [ver]tically bored to provide the journal for the br[ack]et pivot **17**. It will be understood that th[e as]sembly shown in Fig. 4 represents the inb[oard] portion of the surface A as indicated at the l[eft of] Fig. 1. Two bracket arms **13** are provide[d for] each balance surface A and a vertical brack[et] is provided for the support of each bracket **13**.

Rotary pilot forces transmitted through [the] torque shaft **11** are transmitted through [a] sheave or sprocket **20**, and the cable or cha[in] to the sprocket **22**, which is similarly fixed t[o the] outer end of the shorter torque shaft **23**. T[o the] inner end of the latter there is fixed a bevel [gear] **24** which is continually in mesh with the [idler] bevel gear **25** fixed to the upper end of the [pivot] shaft **17**. The bevel gear **25** is in mesh w[ith a] like bevel gear **26** which is keyed or othe[rwise] fixed to the shaft **27**, journalled as at **28** a[nd] within the bracket arm **13**. The outer en[d of] the shaft **27** has keyed thereon a further [bevel] gear **30** which engages a like bevel gear **31** [fixed] to the upper terminal of the outer pivot sha[ft] journalled on a vertical axis within the oute[r end] of the bracket arm **13**. The lower end o[f the] shaft **32** has fixed thereto a control arm **33** [uni]versally connected at its outer terminal t[o the] push-pull rod **34** which in turn is similarly [con]nected to the control horn **35** of the ba[lance] surface A.

It will accordingly be noted that with the [bal]ance surface A in its extended full line po[sition] of Fig. 4, rotation of the torque shaft **11** wil[l im]part rotation in the same direction to the [shaft] **23** and its gear **24** which will cause to r[otate] through the idler gear **25** and the gears at [each] end of the shaft **27**, the gear **31** and its att[ached] vertical shaft **32** to cause rocking of the ba[lance] surface A about its substantially horizontal [pivot] axis A*p*.

Referring to Fig. 5, the conversion unit **16** [com]prises essentially a differential gear assembly [con]sisting of a pair of opposed bevel gears **36** a[nd] **37** rnalled for rotation upon aligned axes and ing a beveled pinion 38 interposed therebe- en and in continual meshing engagement with h of the larger gears. A housing 39 encloses three bevel gears referred to and is provided a hubs or journal portions 40 within which shafts all are adapted to rotate. The hous- 39 is also provided with a radially aligned ring adapted to house the short shaft 41 upon end of which is fixed the bevel pinion 38. forward or opposite end of the stub shaft 41 ttached to a universal joint 42, the forward f of which is internally splined to slidingly age the external splines 33 on the rearward ninal of the torque shaft 15. On the upper tion of the housing 39 there is formed a ket 44 which by means of a clevis connection is pivotally attached to the rear terminal of push-pull rod 14.

ccordingly upon rotation of the torque shaft in either direction the bevel pinion 38 will se the bevel gears 36 and 37 to rotate in op- ite directions causing similar opposite rota- of the shaft portions 11 and 11a to thereby se the mechanism shown in Fig. 4, to provide osite or differential operation of the auxil- balance surfaces A for aileron action. If, ever, it is desired that each of the balance aces A be rocked about their respective hori- tal pivot axes in the same direction, either ardly or downwardly for elevator action, it nly necessary that the pilot prevent rotation he torque shaft 15 and move the push-pull ft 14 in the desired fore and aft direction. gitudinal movement of the shaft 14 causes ing of the housing 39 about the spanwise s of the shafts 11 and 11a, but inasmuch as shaft 15 is prevented from rotating, the bevel on 38 serves as a locking gear to cause the erential gears 36 and 37 to rotate together in same direction with the housing 39 and the fts 11 and 11a. It will be understood that rther universal joint similar to that shown at would be provided in the forward portion of torque shaft 15 to permit this shaft to follow rotary movement of the housing 39, either ardly or downwardly, and to permit the spline to compensate for the variation in distance ween the centers of the respective universal ts.

eferring again to Fig. 4, the mechanism gen- ly designated as 12 for the extension and re- tion of the balance surface A will now be cribed. A motor 18, which may be either of electric, hydraulic or other type, is provided a gear housing 18a and a drive shaft 46 to ch is keyed a worm 47 in engagement with worm gear 48. The latter is journalled upon aforementioned vertical pivot shaft 17 and ixedly attached to rotate with bracket lever through its bolted connections to the lugs 49 reof. A double-arm yoke 50 is pivotally nted and freely rotatable upon the outer ver- l pivot shaft 32 for guided horizontal move- it about its vertical axis within the slotted tion 51 of the bracket arm 13. It will ac- dingly be seen that the pair of bracket arms pivotally interconnecting the rear spar Ws of wing with each pair of yokes 50 forms a par- logram linkage with its corners defined by the s of the vertical pivots 17 and 32. Accord- y as the motor 18 is operated by a suitable pi- control its driven worm 47 imparts rotation to worm gear 48 and outward parallel swing- of the arms 13, the yokes 50 and the attached balance surface A, which is at all times maintained in positions which are parallel to that which it occupies when retracted and nested against the trailing edge of the wing W, while at the same time it is displaced outwardly from the longitudinal plane of symmetry of the airplane. It should also be noted that the surface A is capable of being held and operated in any position intermediate its retracted and extended positions.

It will also be noted that the mechanism for the extension and retraction of the balance surfaces A is independent of the setting or control of the mechanism or the rocking of the surface about its pivot axes Ap journalled within the rearmost portion of the arms of the yoke 50. The control for the motor 18 is, however, preferably interconnected with the control means for the extension and retraction of the flap F in order that both the balance surfaces A and the flaps F be extended and retracted automatically and simultaneously unless such automatic interconnection is deliberately overridden or eliminated by the pilot. It should also be noted that the differential gear mechanism 16 shown in Figure 5 can be operated either for elevator or aileron action of the balance surfaces A regardles of whether the latter surfaces are in their retracted or extended positions. Conversely it will also be apparent that regardless of the position into which these surfaces have been rocked, the extension and retraction mechanism 12 is effective whether selectively controlled by the pilot or automatically actuated by his extension of the flaps F for take-off or landing.

In Figure 6 there is shown a modified form of mechanism for actuating the rocking of the balance surface A wherein cables and sheaves have been substituted for the several bevel gear sets shown in Figure 4. A generally similar bracket arm 52 is pivotally supported for rotation with respect to the bracket 53 supported from the wing structure and carries at its outer recessed portion a yoke 54 supporting the pivotal mounting for the balance surface A. The bracket 53 carries a vertical pivot shaft 55 upon which the arm 52 is adapted to rotate and the latter in turn carries a vertical pivot shaft 56 upon which the yoke 54 is adapted to similarly rotate. It will be understood that suitable mechanism, of which several types are known and available, will be provided to selectively impart movement in either the same or opposite directions to the cables 57, the sheaves 58, and through a continuous cable 59, to the sheaves 60, 61 and 62. These cables are preferably locked to their respective sheaves to insure positive rotation thereof and since the sheave 62 is fixedly attached to the upper terminal of three pivot shafts 56 the desired rotation of control lever 63 is obtained and the locking of the balance surface A is accomplished to the desired extent. The mechanism for projecting the surface A may be similar to that shown in connection with Figure 4.

The improved arrangement and mechanism which has been shown and described herein accordingly provides an advantageous and efficient means for balancing the diving moments which are created, particularly in aircraft of the tailless type, by the extension of the flaps, and the present invention accomplishes these results with mechanisms which are positive acting, of a high strength-to-weight ratio and relatively efficient in their operation and results. Other forms and modifications of the present invention both with respect to the general details of the respective parts are intended to come within the scope and spirit of this invention as more particularly defined in the appended claims.

We claim:

1. In a tail-less airplane a central fuselage, sustaining wings extending laterally from each side of said fuselage, directional control means associated with said sustaining surfaces, high lift flaps associated with the inboard trailing portions of said sustaining surfaces, balance surfaces associated with the trailing portions of said sustaining surfaces outboard of said flaps, means to simultaneously extend said high lift flaps and said balance surfaces into their operating positions rearwardly of said sustaining surfaces and control means for selectively adjusting the angle of attack of said balance surfaces in both their retracted and extended positions.

2. In an aircraft control system, means for extending and supporting a control surface comprising a main wing, a laterally extending rear structural member carried by said wing, laterally spaced pivotal supports carried by said structural member, laterally spaced parallel arms pivotally carried upon said pivotal supports, a yoke pivotally mounted upon the outer end of each said arm having pivotal supports to which said surface is horizontally pivoted, means to rotate said arms for the simultaneous rearward and laterally outward extension of said control surface and means to rotate said control surface in both its retracted and extended positions.

3. In a control surface operating assembly, a main sustaining surface, a control surface disposed adjacent the trailing edge thereof, pivotal supports carried by said main sustaining surface, a pair of arms pivotally mounted upon vertical axes upon said pivotal supports for swinging in substantially horizontal paths, a vertically disposed pivot carried at the free end of each of said arms, a yoke pivotally carried upon said verti arm end pivots for rotation in a horizontal p and having a horizontal pivotal connection at outer terminals for the pivotal support of control surface, means to rotate said arms f aligned spanwise positions adjacent said susta ing surface trailing edge, and control means cluding rotatable transmission elements co-axi mounted upon said vertical pivot axes to said control surface in its retracted and exten position.

HARRY A. SUTTO
ROLF EVERS.

REFERENCES CITED

The following references are of record in file of this patent:

UNITED STATES PATENTS

Number	Name	Date
1,051,429	Merck	Jan. 28,
1,274,986	Carolin	Aug. 6,
1,889,295	Rosatelli	Nov. 29,
1,987,050	Burnelli	Jan. 8,
2,130,958	Kramer	Sept. 20,
2,156,994	Lachmann	May 2,
2,172,289	Munk	Sept. 2,
2,207,453	Blume	July 9,
2,210,642	Thompson	Aug. 6,
2,218,114	Kunze	Oct. 15,
2,218,822	Joyce	Oct. 22,
2,236,838	Robert	Apr. 1,
2,243,885	Schweisch	June 3,
2,246,116	Wagner et al.	June 17,
2,252,656	Youngman	Aug. 12,
2,313,768	Putt	Mar. 16,
2,397,526	Bonbright	Apr. 2,

OTHER REFERENCES

"Aircraft Engineering," February 1945, p 41–45.

PLAINTIFFS' EXHIBIT No. 30

This report constitutes a patent disclosure which Don Hall received 12/20/44. Before docketing this discl. a formal written discl., signed by Dr. Garbell was requested by Don Hall. Garbell was under the impression that this case was under consideration by Pat. Dept. D. A. H.'s request was made to Evers. I didn't know such a case existed.

This case relates to a "method of determining the airfoil sections to be used in new airplanes" and it is questionable whether this is truly an invention and whether it is of a patentable nature. This question would have to be determined first. If it is believed to be of a patentable nature, a signed disclosure should be requested from D. A. H. and the case docketed.

This method of determining the shape of airfoils (at 3 different points along the span) has been used in designing the models 107, 110, XB46 and has been proposed for the model 37.

[In margin]: Date? Addressee?

/s/ STEVE.

Admitted November 24, 1950.

PLAINTIFFS' EXHIBIT No. 31

Consolidated Vultee
Aircraft Corporation
General Offices
San Diego 12, California

March 26, 1947

Mr. Maurice A. Garbell
1714 Lake Street
San Francisco 21, California

Dear Mr. Garbell:

We have completed our investigation of the above referenced disclosure and have decided to inactivate it.

An extensive search of the prior art was made and in our opinion the existing patent art has a very definite limiting effect on the patent coverage that could hope to be secured. We do not feel that the coverage that might be obtained warrants us in prosecuting this disclosure through the United States Patent Office.

CVAC is not utilizing your invention and inquiries to our engineering force indicate that there is no contemplation that it will be used in the future. We have been informed that considerable research by our wind tunnel and aerodynamic groups would be entailed before the utility and efficiency of your construction could be determined, and until such research was performed it would not be considered for use in any of our designs.

Thus since your invention is in the paper stage and there is no use made of it and no contemplated use in mind and the extent of patent coverage doubtful, the Patent Department is inactivating this case.

Very truly yours,

CONSOLIDATED VULTEE
AIRCRAFT CORPORATION

/s/ WALTER J. JASON,
Patent Department.

WJJ/jp

Consolidated Vultee
Aircraft Corporation
General Offices
San Diego 12, California

April 7, 1947

Dr. Maurice A. Garbell
1714 Lake Street
San Francisco 21, California

Dear Dr. Garbell:

In my letter of March 26, 1947, the docket being discussed was inadvertently omitted from the heading of the letter. The reference which was omitted is as follows:

High Speed Air Intake Docket 1129-P.

Yours very truly,

CONSOLIDATED VULTEE
AIRCRAFT CORPORATION,

/s/ WALTER J. JASON,
Patent Department.

WJJ:mm

Admitted November 24, 1950.

PLAINTIFFS' EXHIBIT No. 32

S. D. Dev.

Garbell, M. A. 661-379745 SDD

234-A-22	Slotted Armor Plate	3/30/43	Inactive
301-D-19	Retractable Tail Surfaces	4/ 1/43	Inactive
1129-C	High Speed Air Intake	11/18/44	Inactive
1128-R	Hydrofoil	11/18/44	Inactive
1144-P	Droppable Jet Augmentor	12/ 1/44	Inactive
1237-D	Wing Tip Fin for Tailless Airplane	3/ 1/45	Inactive
1336-D	Longitudinal Control for Jet Aircraft	4/30/45	Inactive
1562-Q	Method of Airfoil Section	1/24/46	Inactive

Admitted November 24, 1950.

DEFENDANTS' EXHIBIT B

(Exhibit 11 (Answer to Interrogatory XXXVI))

Confidential

Consolidated Vultee Aircraft Corporation
Development Engineering, San Diego, Calif.

March 7, 1944

Summary of Wind-Tunnel Tests of a Power-Off 0.058-Scale Model of a Proposed Two-Engine Tailless Design

Preliminary tests of a proposed Two-Engine Tailless Design were made on a 0.058—scale power-off

Defendants' Exhibit B—(Continued)

model in the Galcit 10-foot wind tunnel during the period of February 28 to March 5, 1944.

These tests indicate the revised wing described in Reference 1 is satisfactory from the viewpoint of static longitudinal stability even though the stall, with elevator zero and deflected up for trim at high CL's, and from the viewpoint of elevator effectiveness with flaps up (dCm/dSe = —0.004; see Figure 1).

The characteristics of turbulent separation near and at the stall, as indicated by tuft surveys and three-component force data obtained during the present tests are greatly improved on the new wing over those of the old wing summarized in Reference 2. This is evidenced by:

1) The stall begins at the 35 per cent span point near a CL of 0.9 (elevator zero) and spreads slowly spanwise along the trailing edge (Figure 2).

2) The lift-curve slope is straight up to CL = 0.9, as compared to a separation bend near CL = 0.7 encountered with the old wing.

3) The pitching-moment curve is stable through the stall as compared to the unstable separation kink found in Reference 2. From miscellaneous wind tunnel data on various recent tailless designs it is found that similar desirable stall characteristics are not obtained on other tailless designs with flaps retracted. A comparison report on these data is being prepared.

Figure 1 shows that the relatively enlarged fuse-

Defendants' Exhibit B—(Continued)

lage and nacelles on the new model have a greater destabilizing effect on the static longitudinal stability of the complete configuration than on the old model. Steps are being taken to reduce the fuselage and nacelle overhangs ahead of the wing leading edge, and a modified model will be tested before the conclusion of the present Galcit test period (March 13, 1944).

The first runs of the present test series had indicated an adverse effect of the fuselage on the pitching-moment coefficient at zero lift (in a diving sense) and hence on the trim lift coefficient with zero elevator (toward lower trim). A reduction of the wing incidence at the fuselage from 5° to 2° eliminated this disturbing effect of the fuselage on the trim lift coefficient without any other undesirable consequences, as shown by Figure 3.

The drag of the new model does not differ substantially from that of the previously tested model, as shown in Table 1. It is interesting to note that the value of the span efficiency e is greater on the new model.

Directional stability tests made during the present test period are held in doubt as the fin airfoil section, which is critical for proper stability through zero yaw, has been found to be in error. The San Diego model shop is building a new set of fins with the proper section (NACA 4306), and the new fins will be tested on the revised power-off model and on the power-on model which is in the tunnel now.

Defendants' Exhibit B—(Continued)

References:

1) Report on Selection of Airfoils for the Revised Wing of the Two-Engine Tailless Airplane. C.V.A.C., Dev. Eng., Report ZA-101; February, 1944.

2) Report on Galcit Wind Tunnel Tests of a 0.0639—scale Model of a Two-Engine Tailless Airplane. C.V.A.C., Dev. Eng., Report ZT-021; December, 1943.

By /s/ M. ROGERS,

By /s/ W. E. STROHMEYER,

Checked:

/s/ M. A. GARBELL.

Approved:

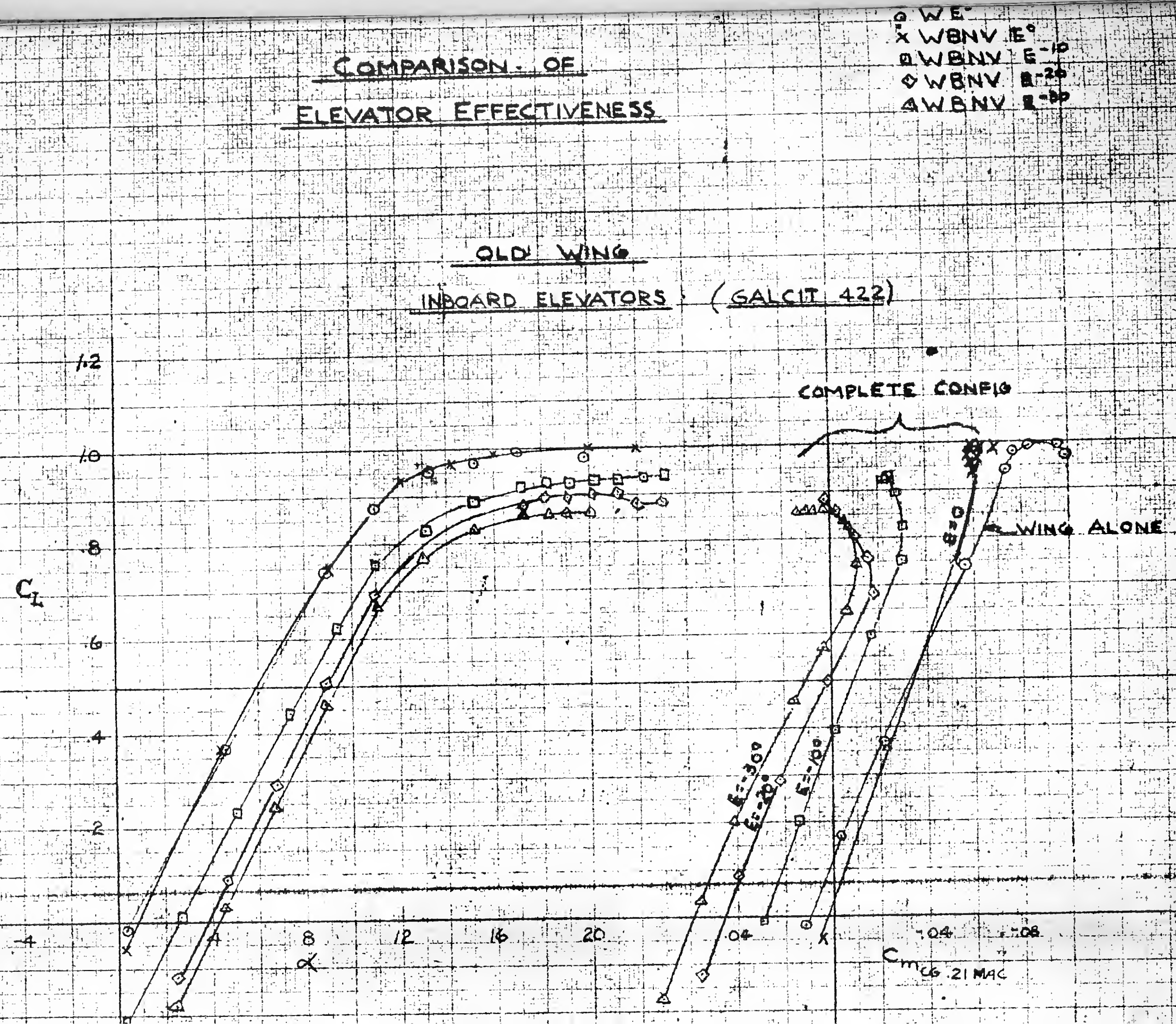
....

Prepared at Galcit.

COMPARISON OF ELEVATOR EFFECTIVENESS

OLD WING

INBOARD ELEVATORS (GALCIT 422)



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SAN DIEGO, CALIF.

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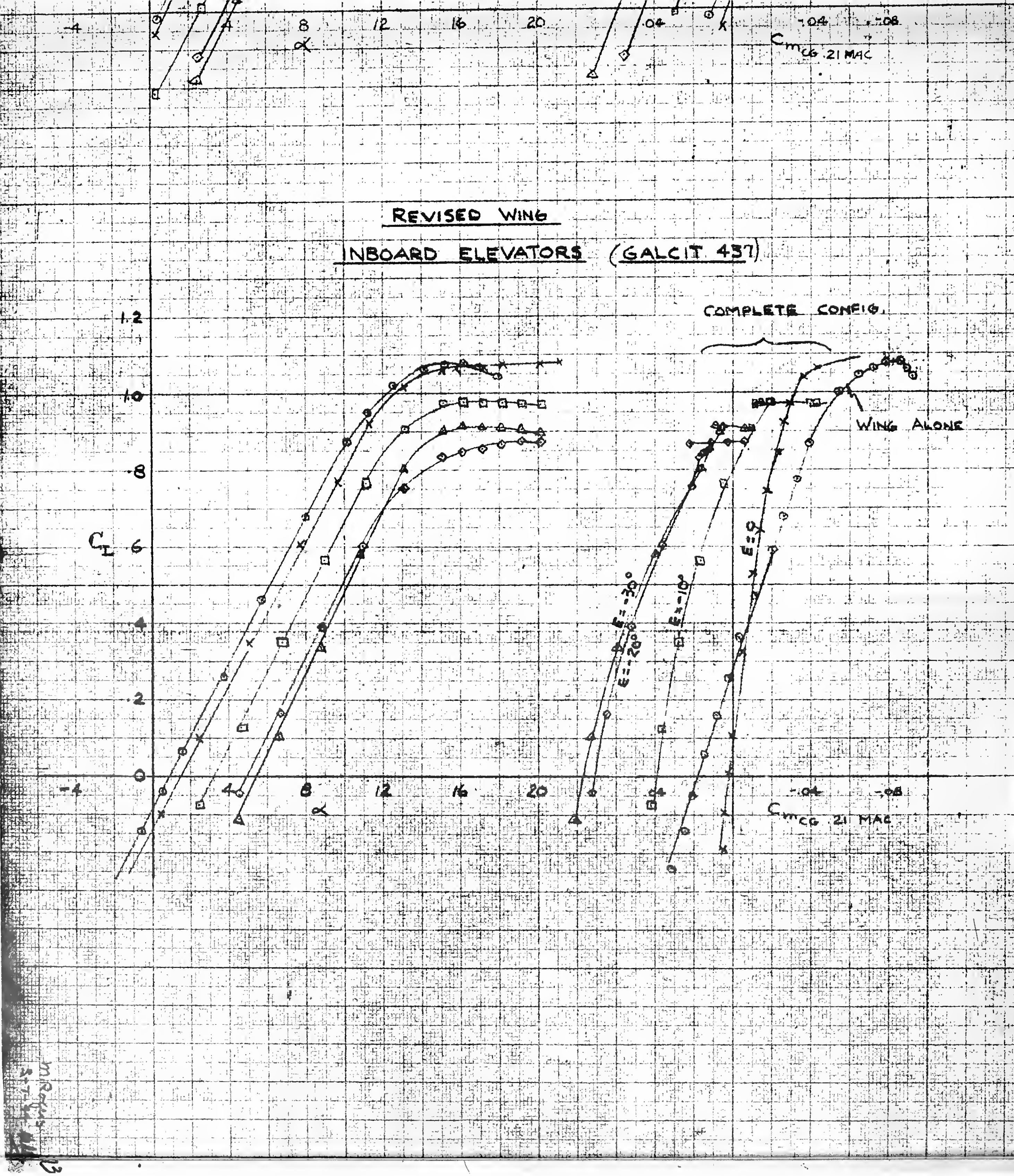
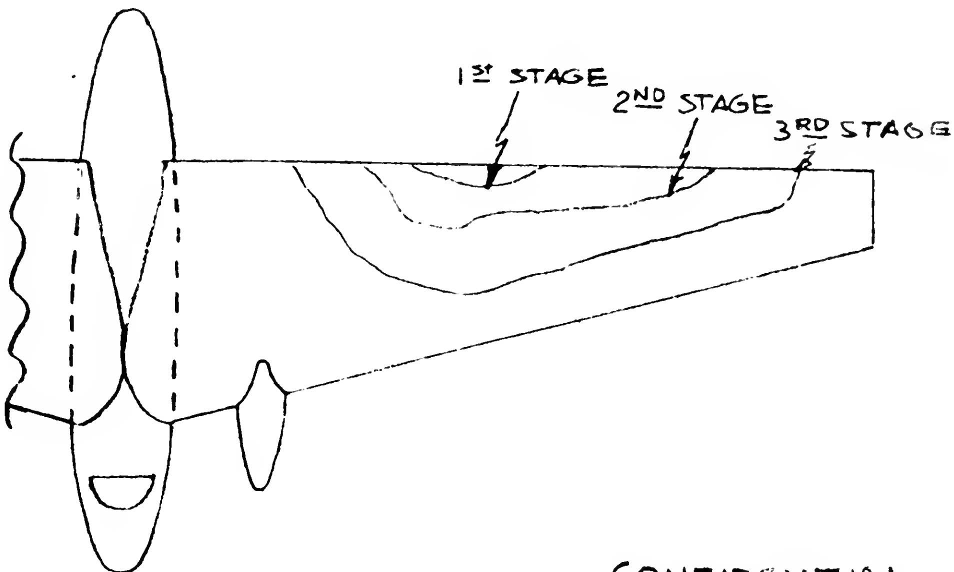


FIG. 1

PAGE OF 8

COMPARISON OF STALL CHARACTERISTICS

FIG. 2



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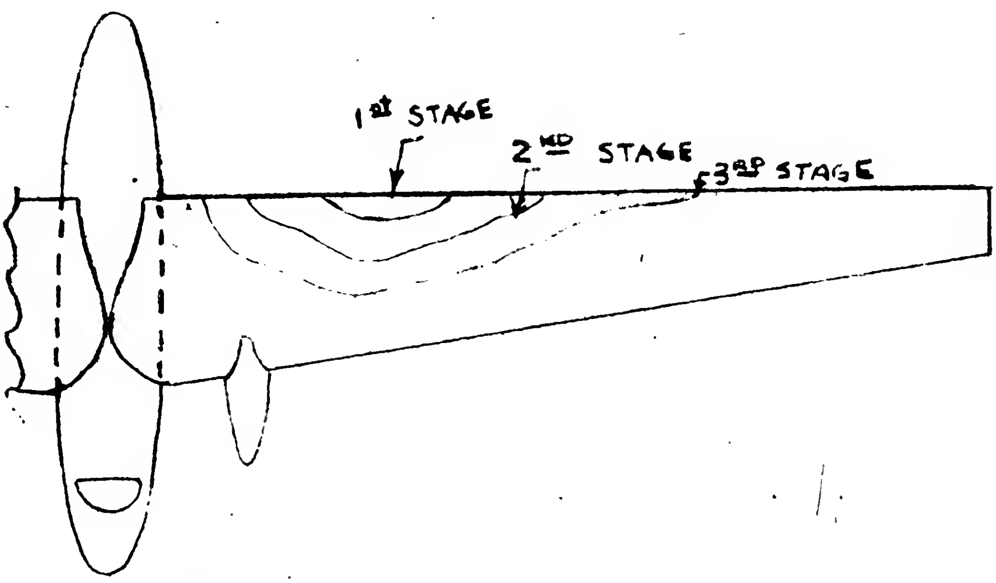
OLD WING

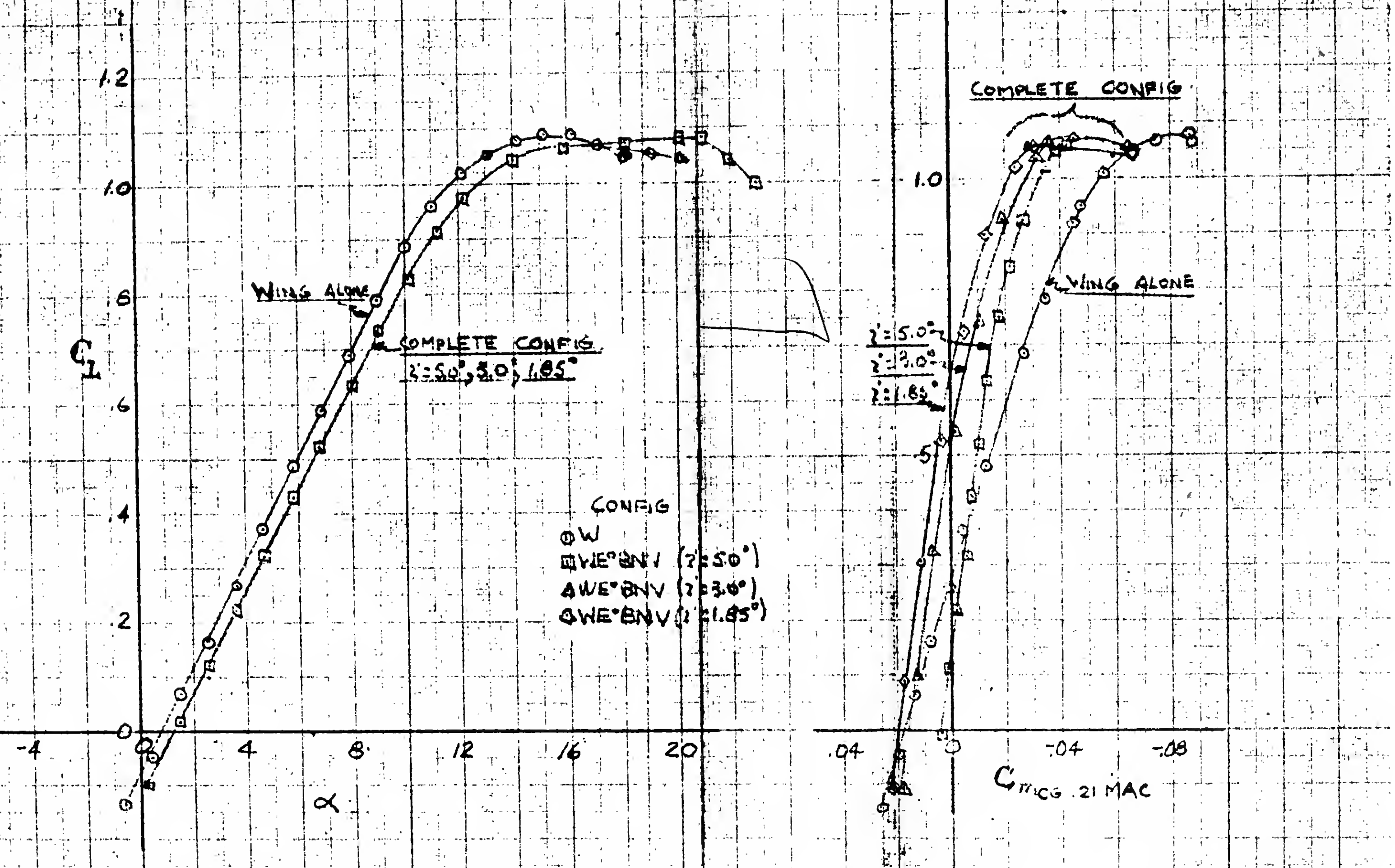


REVISED WING

M. Rogers
3-7-44

FIG 3.

EFFECT OF WING INCIDENCE (GALCIT 437)



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M. Rogers
3-8-44

CONSOLIDATED VULTEE AIRCRAFT CORP.
NEW ENGINEERING SAN DIEGO, CALIF.

FIG. 4

FLAP EFFECTIVENESS (GALCIT 437)

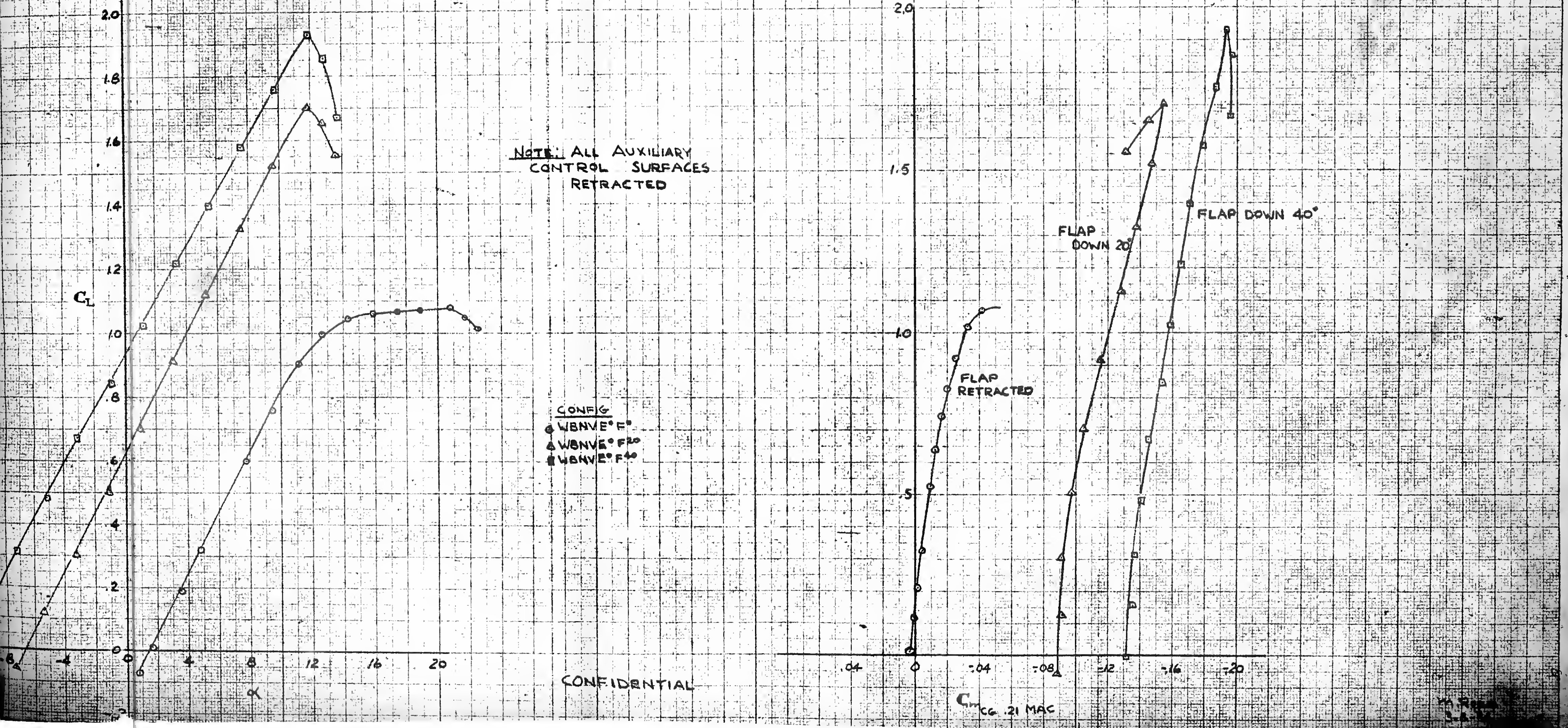

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TABLE I.

SUMMARY OF DRAG VALUES.

	Old model GALCIT 422			Present model GALCIT 437		
Configuration	C_{DP} at $C_L=.3$	C_{DP} at $C_L=.6$	e average	C_{DP} at $C_L=.3$	C_{DP} at $C_L=.5$	e average
W Wing	.0086	.0103	.835	.0087	.0092	.933
WB Wing + Body	-	-	-	.0100	.0105	.935
WBN Wing + Body + Nacelles	.0119	.0133	.855	.0122	.0134	.855
WBNV Wing + Body + Nacelles + Wing-tip fins	.0137	.0142	.945	.0139	.0147	.900

CONSOLIDATED VULTEE AIRCRAFT CORPORATION
GENERAL OFFICES . . SAN DIEGO, CALIFORNIA

PD #400
March 31, 1944

Subject: B-32 Wing Incidence

Reference:
(a) Mr. T. P. Hall's memo #1922, dated March 25, 1944
(b) CVAC Report #ZT-33-001, "Wind Tunnel Test of ZB-32; Part VII; Power test with 5° and 3° wing incidence" February 10, 1941; Ref. Galcit 287-E
(c) CVAC Report T-33-008: "Report on UWAL Wind Tunnel. Test to study flow condition at tail location of the B-32 airplane." October 25, 1943.

In compliance with Reference (a), the following information on the history of the wing incidence on the subject airplane has been compiled.

The original decision regarding the wing incidence was based on the conclusions of Reference (b). The two criteria for the decision to use 3° incidence were static longitudinal stability with power on and drag. The following summarized values indicate the effect of wing incidence on these two items:

Wing Incidence	Static Long. Stability (C.G. = .25 MAC)		Parasite Drag C_{D_P}	
	Power off	Rated Power	C_L = .3 High spd.	C_L = .75 Cruise
3° (Runs 121 and 127 of Ref. (b))	-.17	-.16	.0299	.0330
5° (Runs 7 and 125 of Ref. (b))	-.18	-.14	.0302	.0333

The increase in drag was considered acceptable; however, the 3% greater destabilizing effect of power was considered prohibitive.

Wake surveys presented in Reference (c) justified the choice of the 3° incidence. In this reference it is concluded that, in order to avoid buffeting "the lowest tail position practicable is recommended". A change of wing incidence from 3° to 5° will effectively raise the tail 11 1[illegible] inches, and increase the extent of the buffeting range.

HAS requested we show conditions for which drag values are listed
i.e. TRIM FLAP ETC.

By Magarbell

Checked Bayless

Defendants' Exhibit B—(Continued)

Consolidated Vultee Aircraft Corporation
General Offices San Diego, California

Aero Memo—#238
April 13, 1944.

Mr. T. P. Hall
Mr. M. A. Garbell

Wind Tunnel Tests of Two-Engine Tailless Airplane at M.I.T.

Enclosure: (A) Three (3) Plots of Preliminary Wind Tunnel Data.

The power-off tests of the two-engine tailless airplane were conducted from March 28 to April 7. The wing was modified from the wing tested in Galcit. The present wing has 14° leading edge sweepback as against the 11° and 15° sweepback previously tested during the early part of March. The fuselage was shortened and the wing incidence set at 2°.

The elevator effectiveness and stall characteristics with flaps up (Fig. 1) are impared by the faulty model (there is a slight difference in wing incidence as borne out by unsymmetrical stall patterns) and by the angularity of the wing airfoils caused by the rotation of the wing about an arbitrary point on the root chord to obtain the desired sweepback. The effectiveness of the outboard elevators (extending from the outboard end of the present inboard elevators to the wing tip) is almost twice that of the retracted high-aspect ratio aft surface elevators.

The elevator control available at the stall with

Defendants' Exhibit B—(Continued)

flaps extended is shown in Figures 2 and 3. The original high aspect-ratio aft surface (A=17) was tested in two horizontal positions along the root chord line (142 inches and 166 inches aft of the retracted position). At the shorter tail position, the lower elevator effectiveness was offset by the reduced static longitudinal stability so that elevator control, with either configuration, is powerful enough to stall the airplane. However, both horizontal positions of the high-aspect ratio tail show the same tail stall and marginal longitudinal stability at low lift coefficients with large up elevator deflections.

A lower aspect-ratio tail of 7, with approximately 20% more area, was tested 182 inches aft of its retracted position (Fig. 3). The elevator control is powerful enough to stall the airplane at CL max= 2.3 (full scale) and the tail stall experienced with the high-aspect ratio tail is eliminated.

M. A. GARBELL.

MR:EML

cc: Dev. Engr. File
Aerodynamics Ofc. #16

M.I.T. TESTS
2-ENGINE TAILLESS DESIGN
MAR.-APRIL 1944

PRELIMINARY DATA

FIG. 1
ELEVATOR EFFECTIVENESS
FLAPS UP

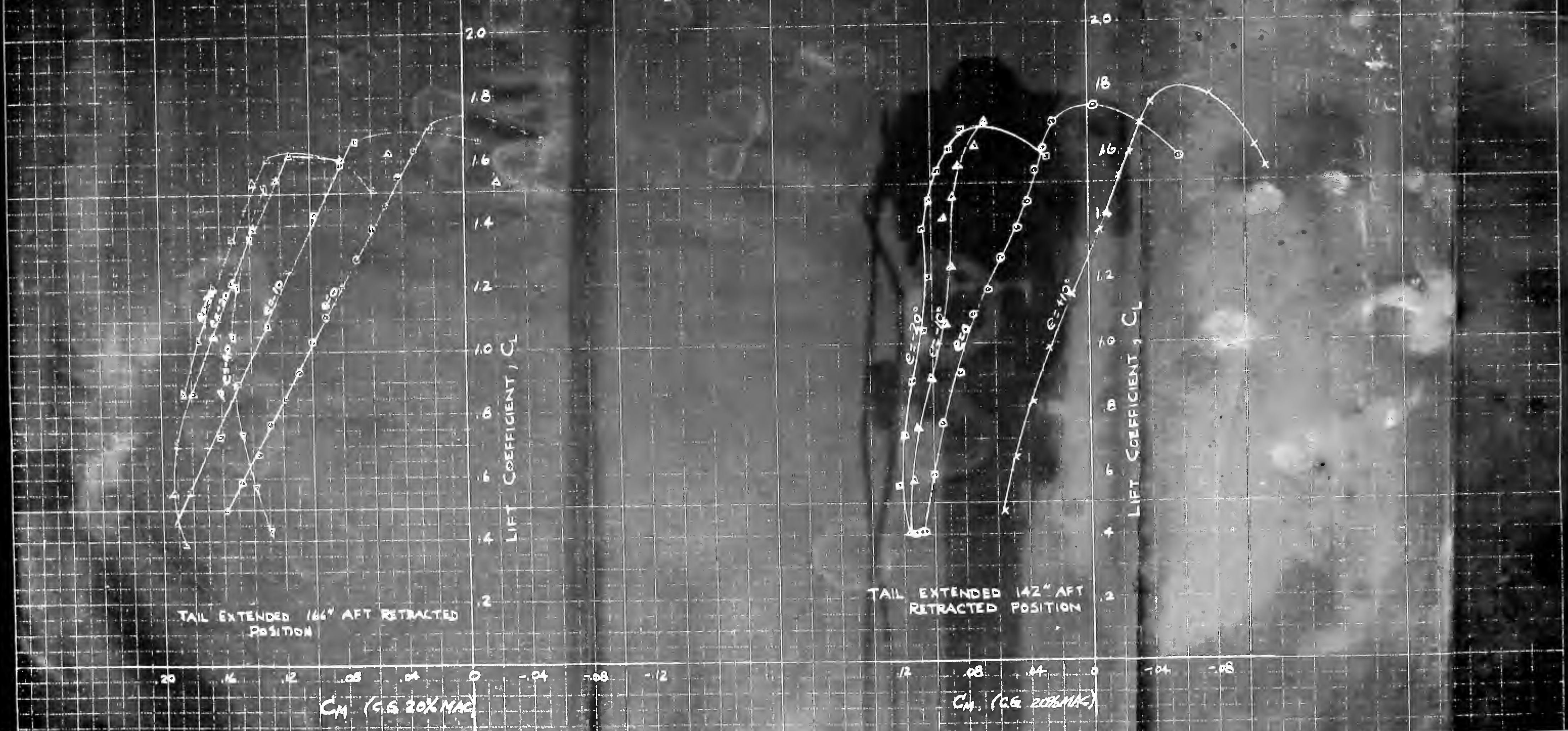
COMPLETE MODEL

M.I.T. TESTS
2-ENGINE TAILLESS DESIGN
MAR.-APRIL 1944

PRELIMINARY DATA

237

FIG. 2
ELEVATOR EFFECTIVENESS
FLAPS DOWN 40°
COMPLETE MODEL - TAIL ASPECT RATIO = 1.7
q = 50 lb/ft²



81

M.I.T. TESTS
ENGINE TAILLESS DESIGN
MAR-APRIL 1944

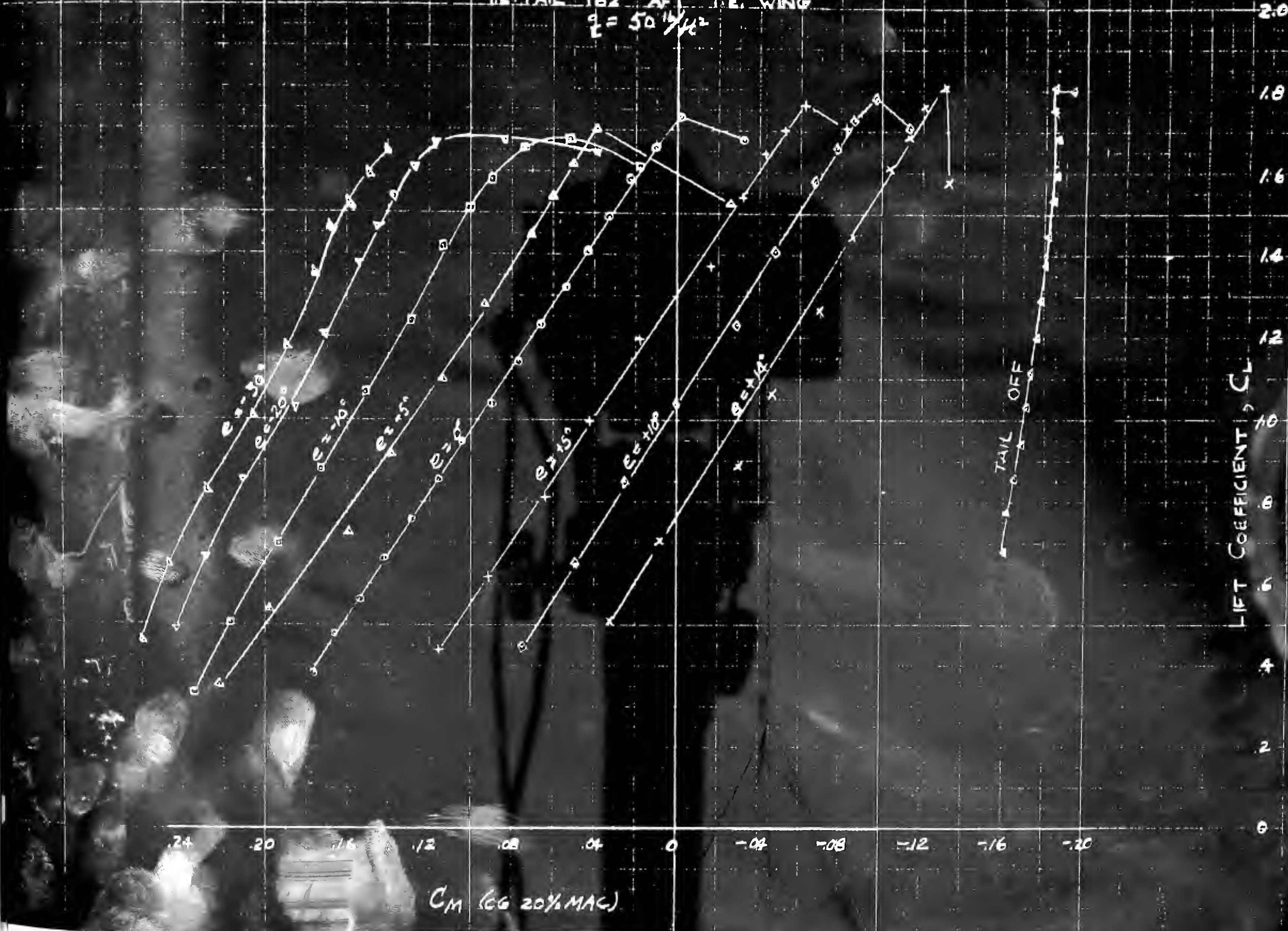
PRELIMINARY DATA

238
9

FIG. 3

ELEVATOR EFFECTIVENESS
FLAPS DOWN 40°

COMPLETE MODEL - TAIL ASPECT RATIO = 7
T.E. TAIL 182" AFT T.E. WING
$q = 50\ ^{lb}/_{ft^2}$



Defendants' Exhibit B—(Continued)

Consolidated Vultee Aircraft Corporation
General Offices San Diego, California

Aero Memo #250
April 15, 1944

Mr. M. Rosenbaum
Mr. M. A. Garbell

Longitudinal Stability and Control Data for Structures. XB-32 Airplane with B-29 Single Tail Installation.

(a) A.V.C. from M. Rosenbaum to C. Blake dated March 10, 1944.

(b) Aero Memo #206 dated March 23, 1944.

In accordance with your request, reference (a), and superseding the data given in reference (b), aerodynamic data for the XB-32 airplane with the B-29 single tail installation is presented in the enclosed table. All data were estimated from the last Galcit test of the airplane, as little flight test data are available.

M. A. GARBELL.

cc. Dev. Engr. File
Aerodynamics Ofc. #16

Defendants' Exhibit B—(Continued)

Consolidated Vultee Aircraft Corporation
General Offices San Diego, California

Aero Memo #260
April 19, 1944.

Mr. C. B. Carroll
Mr. M. A. Garbell

B-32 Intercooler Exit Flap and Effect on Tail Buffeting.

Enclosure (A) One (1) copy Intercooler Air Spillway Installation on B-32 Airplane drawing.

The attached figure shows the approximate relative location of the proposed intercooler flap installation on the B-32 and the present installation on the XB-32. It is expected that the introduction of the intercooler air into the upper portion of the wing wake, together with the disturbance caused by the exit flap, will intensify the tail shake to a similar degree as the upper engine cowl flaps.

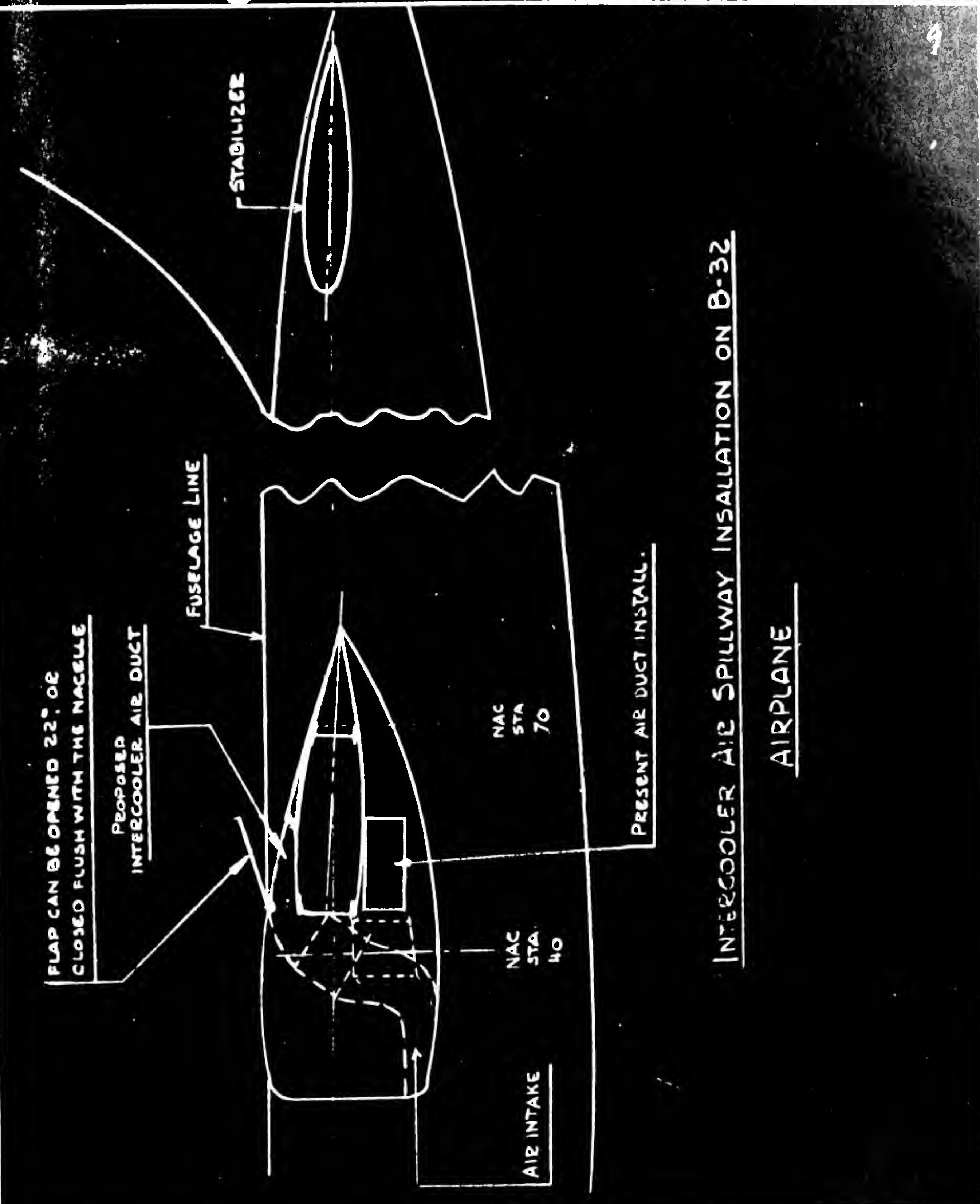
It is believed that, from the standpoint of tail shake, the XB-32 intercooler exit arrangement is preferable.

M. A. GARBELL.

WSS:EML

cc: Dev. Engr. File
Aerodynamics Ofc. #16

9



DRAWN	SKOBLIN	4-15-44	INTERCOOLER AIR SPILLWAY INSTALLATION ON B-32 AIRPLANE	PART NUMBER
APPROVED				
APPROVED			CONSOLIDATED AIRCRAFT CORPORATION LINDBERGH FIELD . SAN DIEGO, CALIF.	

CONSOLIDATED VULTEE AIRCRAFT CORPORATION
GENERAL OFFICES . . SAN DIEGO, CALIFORNIA

Aero #261
April 19, 1944

WIND-TUNNEL TEST OF THE SKYCOACH

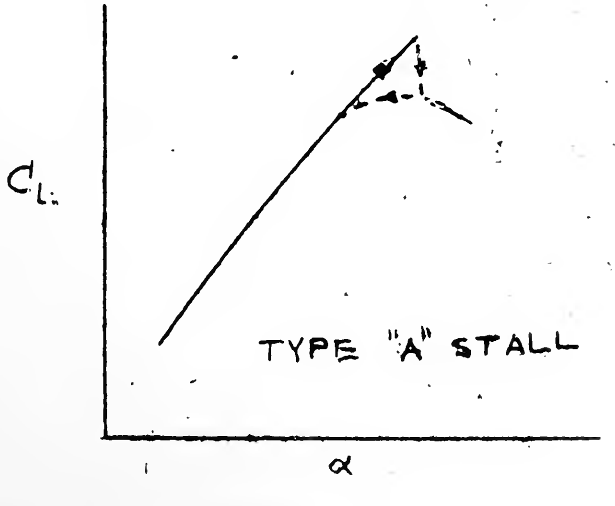
Reference: (a) Mr. A.G. Tsonga's Memo to Mr. T.P. Hall, dated April 8, 1944
(b) Engineering Report No. 1486, CVAC Stinson Division dated Dec. 31, 1943

The suggestion to change the geometric washout of the Skycoach model wing from 0° to 3° (contained in Reference (a)) has been studied, and the following comments are made:

1. The original washout distribution as shown on page 5 of Ref. 2 had zero washout from the wing root to the tail boom juncture and 1.9° washout at the wing tip. Our earlier studies of the Skycoach showed that this wing design would have satisfactory stalling characteristics. The wing stall with this washout distribution and the latest planform should start at the booms. The outboard 30 percent wing-span portion of the wing containing the ailerons should remain unstalled until the flow over the entire inboard portion of the wing is stalled.

2. The change to 0° washout in the present design is not understood. We were informed of this change only when the model drawings arrived here for the construction of the model. The stalling characteristics of this wing are anticipated to be somewhat unfavorable. The stall will probably start simultaneously at the tail boom-wing juncture and the inboard end of the ailerons and will spread evenly toward the wing root and tip. It is believed that the washout distribution of the original design should be used.

3. A further study of the airfoils to be employed shows the questionable value of the five-digit airfoils proposed for the Skycoach. Both the root airfoil (23015) and the tip airfoil (43012-A) have stall characteristics of the type "A" shown below, that is, have different stalling and unstalling lift curve peaks, as shown.

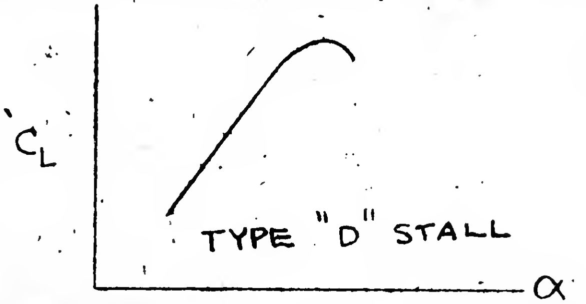


CONSOLIDATED VULTEE AIRCRAFT CORPORATION
GENERAL OFFICES . . SAN DIEGO, CALIFORNIA

Aero #261
4-19-44

If the airplane is brought to a stall, a temporary premature separation will make one wing follow the lower stalled lift curve (see above sketch) while the other wing follows the upper unstalled lift curve. A decided dive is then necessary to put an end to the ensuing rolling moment, as corrective aileron action contributes only to aggravate the unsymmetric stall and the auto-rotative tendency of the airplane.

The use of the NACA 2415 section at the wing root and the 4412 section at the tip would improve the stalling characteristics and the handling characteristics at the stall substantially, because both airfoils have a smooth "D" type stall, free of any unstalling hysteresis, as shown below:



The geometric washout with these four-digit airfoils, because of the greater difference in zero lift angles, should be 3° at the tip with zero washout at the wing boom juncture.

The increase of drag of about $\Delta C_{D_p} = .0015$ as obtained from NACA T.R.'s 460 and 661, caused by a change from the present five-digit airfoils to the more desirable four-digit airfoils is purely fictitious inasmuch as the greater sensitivity of the five-digit airfoils to surface roughness equalizes the drag of two wings of comparable normal manufacturing quality. The theoretical loss in section $C_{L\max}$ of about .15 is also not believed to be representative of the actual $C_{L\max}$ of the airplane because the tail booms have a greater detrimental effect on the wing stall on the five-digit airfoils (as evidenced by the tuft photos in Reference (b)) than would be the case on a four-digit airfoil wing.

The writer made a direct comparison of a five-digit wing and a (2415-4409) wing on the same type high-performance sailplane in 1937. The results as observed and measured in flight confirmed fully the above considerations. Another example of somewhat undesirable handling characteristics at the stall due to lift hysteresis is the DC-3.

4. A second wing block for a revised wing is ready in the Model Shop. It is suggested that a revised wing be built and tested.

MAGarbell

Defendants' Exhibit B—(Continued)

Consolidated Vultee Aircraft Corporation
General Offices San Diego, California

Aero Memo #278
May 3, 1944

Mr. T. P. Hall
Mr. M. A. Garbell

Camber in Horizontal Stabilizer—B-32 Airplane.

(a) Memo # 1955 to R. L. Bayless from T. P. Hall dated April 3, 1944.

(b) Aero Memo # 188 to T. P. Hall from C. L. Blake dated February 25, 1944.

(c) Memo to R. C. Sebold from R. H. Widmer dated March 23, 1944.

Enclosure: (A) Doc. Aero 33-107. Revised May 1, 1944. Plot of elevator deflection for trim versus center of gravity position.

The change in camber of the horizontal stabilizer from a negative cambered section which is now on the airplane to a symmetrical section will decrease the down elevator required to trim by approximately 0.8°. This value is in agreement with Widmer's data quoted in Reference (c) when considered in terms of the effective change in stabilizer incidence.

The second paragraph of Reference (c) states that the Boeing horizontal surface was cambered to prevent the lower surface from stalling with flaps fully deflected at low lift coefficients. Our wind-tunnel data on the XB-32 with the cambered Boeing surface and with our earlier symmetrical surface show no stall even down to negative lift

Defendants' Exhibit B—(Continued)

coefficients. Therefore, reforming the stabilizer nose to give a symmetrical section is considered to be permissible on our airplanes.

Enclosure (A) is similar to the chart included with reference (b) except that the CL for start of long range flight was changed from 0.9 to 0.85 to agree with recent information received from Fort Worth. Also the velocities corresponding to different gross weights and lift coefficients have been added to the original chart.

/s/ J. E. A., for
M. A. GARBELL.

VHG:dh

cc: C. B. Carroll
J. B. Jewell
Aerodynamics (2)
Dev. Engr. File


MODEL XB-32 WITH B-29 SINGLE TAIL INSTALLATION
ELEVATOR DEFLECTION FOR TRIM
VS
CENTER OF GRAVITY POSITION
STABILIZER SETTING = -4°
PRESENT HORIZONTAL TAIL (HAS NEGATIVE CAMBER FWD OF FRONT SPAR)
DATA BASED FLIGHT TEST RESULTS - XB-32
PRESENT HORIZONTAL TAIL WITH NEGATIVE CAMBER REMOVED (SYMMETRICAL).
ESTIMATED DATA
MOST PROBABLE AFT C.G. FOR PRODUCTION B-32.
PRODUCTION B-32 IN CRUISING FLIGHT.
START OF LONG RANGE FLIGHT. G.W. - 100,000# 120,000# V - 269 M.P.H. 293 M.P.H. ALT : 25,000 FT.
END OF LONG RANGE FLIGHT G.W. - 65,500# V - 230 M.P.H. ALT : 25,000 FT.
PRODUCTION B-32
HIGH SPEED
MOST PROBABLE FORWARD C.G. FOR PRODUCTION B-32
CENTER OF GRAVITY POSITION - % M.A.C.
ELEVATOR DEFLECTION FOR TRIM - DEG.
UP
DOWN
CALCULATED BY
TRACED BY
APPROVED BY
REV.
5/11/44
MODEL XB-32 WITH B-29 SINGLE TAIL INSTALLATION - ELEVATOR DEFLECTION FOR TRIM VS CENTER OF GRAVITY POSITION
CONSOLIDATED AIRCRAFT CORPORATION
LINDBERGH FIELD, SAN DIEGO, CALIF.
PAGE
DOC. NO.
MODEL
XB-32

INTRA-COMPANY CORRESPONDENCE

CONSOLIDATED VULTEE AIRCRAFT CORPORATION
GENERAL OFFICES . . SAN DIEGO, CALIFORNIA

Aero Memo #320
DATE May 22, 1944.

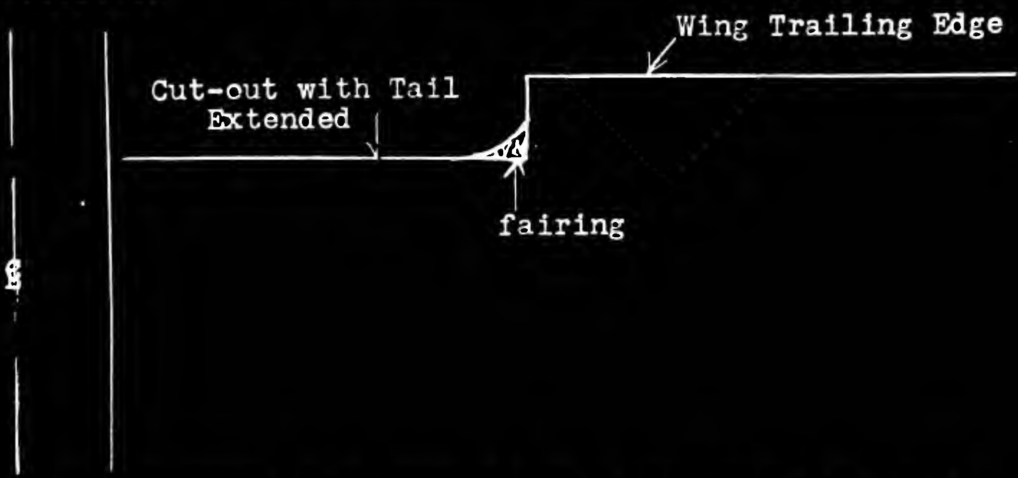
TO Mr. M. A. Garbell

FROM Mr. R. L. Bayless

SUBJECT Two-Engine Tailless Wing Fairing

REFERENCE

Mr. Sutton suggested we fair wing cut-out on tailless as follows:



Please work with Preliminary Design on this and include in next wind tunnel model if feasible.

R. L. Bayless.

RLB:EML

MODEL AIRPLANE REPORT NO.

STINSON SKYCOACH

PRELIMINARY REPORT (IV) ON TESTS

M.I.T.

1/5 SCALE WIND TUNNEL MODEL

JUNE 8, 1944.

The fairly satisfactory wing-fuselage plasticene fillet, which as developed during the past two days, has been replaced by a more urable wood fillet. Most of the abbreviated schedule has now been ompleted.

The maximum lift coefficients

$C_{L_{max}} = 1.25$ Flaps up

$C_{L_{max}} = 1.79$ Flaps deflected 30°

$C_{L_{max}} = 1.84$ Flaps deflected 50°

ndicate normal flap effectiveness except for 50° deflection. Addi- ional future research and testing will be required to obtain a etter flap effectiveness at large angles.

The aileron effectiveness, flaps up, is adequate to give a elix angle $pb/2v = 0.085$. There is no loss in aileron effectiveness to the total wing stall.

The static directional stability after the installation of -54 type dorsal fins is satisfactory through the entire range of awing angles. The numerical value of the directional stability erivative is $dC_n/d\psi = -.0020$. There is no rudder stall up to he maximum rudder deflection of 20°.

Other data are still being computed.

The Stinson test should be completed today with the remaining ower runs for the three flap configurations.

In compliance with Mr. Sutton's request a few runs will also made to obtain constant trim C_L with the various flap deflections.

The subsequent brief tests of the Tailless Model are intended investigate a 6° wing incidence, rolling control effectiveness th the new aileron-spoiler combination (designed to give rolling ntrol without any pitching moment disturbance), and additional oblems of the extended aft surface. Our tests are scheduled to d on Saturday, June 10, 1944.

MAG

BY

CHECKED

APPROVED

92

MODEL AIRPLANE REPORT NO.

STINSON SKYCOACH

PRELIMINARY REPORT (III) ON TESTS

M.I.T.

1/5 SCALE WIND TUNNEL MODEL

JUNE 7, 1944.

A continued enlargement of the aft wing-fuselage fillet did not improve the critical wing root stall any further. Careful observation of the tuft pattern near the wing leading edge, subsequently, lead to the conclusion that the basic reason for the premature flow separation consisted in the critical sensitivity of the airfoil leading edge to the unfavorable pressure distribution caused by the fuselage intersection. A fairly large leading-edge fillet, combined with the original small aft fillet delayed the undesirable wing-root separation to the angle of attack for the maximum lift coefficient ($C_{L\ max.} = 1.25$).

The installation of small dorsal fins on the vertical surfaces straightened the yawing moment curve up to the highest angles of yaw tested (21^{o}).

The attached abbreviated test schedule is being run at present to obtain complete information on the cleaned-up configuration with flaps retracted, partly and fully extended, power-off and with rated power.

At the completion of this schedule, probably Thursday afternoon, the Tailless Model will enter the tunnel for about four days' testing.

M.A.G.

BY

CHECKED

Attachment to:

STINSON SKYCOACH

PRELIMINARY REPORT (III) ON TESTS

M.I.T.

1/5 SCALE WIND TUNNEL MODEL

JUNE 7-8, 1944

1. Flaps up - Power-off and Rated power.

   (a) MDP_6 (power runs are P_6) $e = +10^\circ, 0^\circ, -10^\circ, -20^\circ$ (stabilizer set to trim at $C_L = .3$ with e = (

   (b) Y_6 $e = 0^\circ$ $r = 0^\circ, 10^\circ, 20^\circ$

   (c) P_6 $a = \pm 20^\circ$

2. Flaps 25°-25° - power off and rated power

   (a) P_6 $e = 0^\circ, -10^\circ\ -20^\circ$

   (b) Y_6 $e = 0^\circ,$ $r = 0^\circ, 10^\circ, 20^\circ$

3. Flaps 50° -50° - power off and rated power

   (a) P_6 $e = 0^\circ, -10^\circ, -20^\circ$

   (b) Y_6 $e = 0^\circ,$ $r = 0^\circ, 10^\circ, 20^\circ$

   (c) P_6 $a = \pm 20^\circ$

Note: Runs (a) yield information on static longitudinal stability and elevator effectiveness.

Runs (b) indicate static directional and rolling stability and rudder effectiveness.

Runs (c) together with one run of series (a) give aileron effectiveness.

MODEL AIRPLANE REPORT NO.

STINSON SKYCOACH

PRELIMINARY REPORT (II) ON TESTS AT

M.I.T.
1/5 SCALE WIND TUNNEL MODEL

JUNE 6, 1944

Most of the running time, during the past two days, was edicated to the improvement of the objectionable wing root tall. Although the enlarged fillets raised the break of the ift curve, power-off, from $C_L = .85$ to $C_L = 1.10$, the final reakdown of the airflow over the wing root could not be avoided. nall changes in the fillet and the installation of a small orsal fin on the fuselage top would shift the root stall from ne wing to the other, but in any case the sudden local stall ould cover a comparatively large area.

Attempts were also made to improve the flap effectiveness hich showed an increase in $C_{L_{max}}$ from 1.40 (flaps up) to only .67 (flaps 29°) and a lift decrease with further flap deflec- ion. Changes of the flap gap did not show any appreciable ain in $C_{L_{max}}$.

Dorsal fins similar to those employed on the XP-54 are eing tested today in an effort to improve directional stability large angles of yaw.

MAF

ne 6, 1944.

BY

CHECKED

APPROVED

Defendants' Exhibit B—(Continued)

Consolidated Vultee Aircraft Corporation
General Offices San Diego, California

July 1, 1944

Mr. T. P. Hall
Mr. M. A. Garbell

Free-Flight Tests of Two-Engine Tailless Design

The following is a summary of a telephone conversation between Mr. Shortal, of the N.A.C.A., and Mr. Rogers, of the Aerodynamics Group of C.V.A.C., held June 30, 1944.

1. Free-Flight Tunnel film of the Aspect Ratio 12, Tailless Flaps-Up model, in flight, is now being reviewed at The Bureau of Aeronautics. A copy should arrive in San Diego sometime during the latter part of next week.

2. Preliminary data on the dynamic damping derivatives obtained experimentally on the original, Aspect Ratio 10 Tailless design, have already been forwarded to this company. These data were discussed with Mr. Rogers on his recent visit to Langley Field and show good correlation with the theoretical values given in C.V.A.C. Report ZA-095 on the dynamic stability of the Two-Engine Tailless Design.

3. The flaps-down model of the Aspect Ratio 12 design arrived at the N.A.C.A. in good condition. Force tests on the six component balance have already been made. At present, tuft surveys of the model are being made. The model should be flown sometime during the middle part of next week (about July 5, 1944).

4. Mr. Shortal suggested that, in view of our

Defendants' Exhibit B—(Continued)

interest in aileron-spoiler combinations and the general interest of the aeronautical industry in such data, it may be possible for the Free-Flight Tunnel to run a series of research tests to determine the time response of an airplane with this lateral control system, as well as general flight characteristics. To help him get authorization for such a general research program, Mr. Shortal suggested that this company write a letter to Dr. Lewis of the N.A.C.A. recommending that such a program be undertaken by the N.A.C.A. It is felt that owing to the basic nature of such data it may be possible for the N.A.C.A. to initiate such a program should some manufacturer request information or data of such general interest to the industry.

5. Mr. Shortal again will try to send us some Free-Flight film on the flights of another tailless design, either a basic N.A.C.A. research model or the Kaiser-Koppen Design. Permission to send us this film previously was not granted by the N.A.C.A. on the grounds that they, in all fairness to the rest of the industry, would also have to send the film to all other manufacturers. However, Mr. Shortal feels that a short term loan of the film might be arranged.

M. A. GARBELL.

MR:ms

cc: Aero. File (3), Dev. Engr. File
[In margin]: Filed, Hall.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION
GENERAL OFFICES . . SAN DIEGO, CALIFORNIA

Aero Doc. #Misc.-120

July 5, 1944

Subject: Recommended Design Modifications to Single Engine Pusher Design.

Reference: (a) CVAC Report #ZA-030 - Wind Tunnel Test of a 1/5 Scale Powered Model, Single Engine Pusher Design.

Enclosure: (A) Sketch of present and proposed Flap Slot and Path
(B) Three view of Single Engine Pusher Design with recommended modifications.
(C) Sketch of engine air intake

The following modifications to the single engine pusher design are recommended on the basis of the M.I.T. wind tunnel test summarized in reference (a):

1. The flap slot and flap path should be modified, as indicated by enclosure (A), in order to obtain a maximum lift increment of at least $\Delta C_{L_{max}} = 0.30$ between the 25° and the 50° flap deflection. Only $\Delta C_{L_{max}} = 0.10$ was obtained in the test. The slot and path used on the model of reference (a) are those designed by the N.A.C.A. for use on the 23012 airfoil, and they are not suitable for the 23018 airfoil used on the design. The flap slot and optimum path shown by enclosure (A) are derived from the configuration 2(b) of N.A.C.A. T.R. 677, which was originally designed for the 23021 airfoil and which is believed to be equally effective for the 23018 airfoil.

2. The tail length should be increased approximately 27 inches (15% increase in tail length) and the horizontal tail chord increased 7 inches (15% increase in tail area) to give adequate longitudinal stability at the probable most aft C.G. of approximately 32% M.A.C.* The vertical tail area may be decreased 15% with this increase in tail length as the present directional stability and control are considered satisfactory.

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*Note:
The probable most aft C.G. of 32% is based on Drawing S-43-045 which shows a design rearward C.G. of 28.2% M.A.C. It is estimated that the C.G. will move aft to approximately 30% M.A.C. with a light load and a light pilot (90-100 lb.) for the present design. This figure cannot be accurately determined due to lack of data, but appears reasonable based on earlier studies summarized in Report ZA-099. The increase in tail length and tail modification will result in a C.G. shift aft of approximately 2% M.A.C. due to the increase in weight moment. The resulting most aft C.G. is therefore 32% M.A.C.

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CONSOLIDATED VULTEE AIRCRAFT CORPORATION
GENERAL OFFICES . . SAN DIEGO, CALIFORNIA

Aero Doc. #Misc.-120

July 5, 1944

The criteria for satisfactory longitudinal stability are based on the following data:

$\frac{dC_{M_H}}{dC_L}$ as tested = - .245 (C.G. 25%)

$\frac{dC_{M_H}}{dC_L}$ (increased tail length and area)

= -.245 x 1.15 x 1.15 = - .325

$\Delta \frac{dC_m}{dC_L}$ (C.G. 25%)

(a) C.G. to 32% MAC	+ .07
(b) Power on	+ .02 (test)
(c) Free elevator	+ .04 (estimated)
(d) Airplane tail off	+ .145
Total	+ .275

$\frac{dC_m}{dC_L}$ (C.G. 32%), power on =

= -.325 + .275 = - .05

This margin of static stability is considered adequate for satisfactory flight characteristics.

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3. Dorsal fins, similar to those used in the wind tunnel tests to eliminate vertical surface stall at angles of yaw greater than 5°, should be incorporated in the design (see encl. (B)).

4. The leading edge fillet, used in the wind tunnel tests to obtain reasonably good aerodynamic characteristics, is not a very satisfactory solution to the wing fuselage interference and premature root stall problem as described in reference (a). It is possible that the engine cooling air intake could be moved from its present position at the top of the fuselage to two side ducts in the vicinity of the flow separation at the wing-fuselage intersection (approximately 30% wing chord). This should relieve the unsatisfactory root stall by

CONSOLIDATED VULTEE AIRCRAFT CORPORATION
GENERAL OFFICES . . SAN DIEGO, CALIFORNIA

Aero Doc. #Misc.-120

July 5, 1944

removal of the boundary layer. If the exit air was expelled at the end of the fuselage fairing below the drive shaft to relieve the bluntness of the desired fuselage fairing it appears that the ducting arrangement would be unsatisfactory, and a special fan, now provided on the propeller shaft, would be required at this exit. (See Encl. (C). If the air was expelled around the propeller spinner, as now planned, the fan would absorb greater power than at present, as the duct entrance would be at a lower pressure than the duct exit. Insufficient airplane design details are available at San Diego to investigate this modified duct arrangement. Also, air expelled below the drive shaft would probably cause objectionable interference with the propeller.

This root stall condition could probably be relieved also by use of less critical wing airfoils similar to the NACA four digit series airfoils (i.e. 2518 root and 4412 tip as compared to the present 23018 root and 43012-A tip). Although no pressure distribution data are available for the four digit series airfoils, these airfoils basically have lower peak pressures due to the further aft position of the maximum camber point. Therefore, they should be less sensitive to wing-fuselage interference. However, a trailing-edge fillet will still be necessary to relieve the fast expansion along the rear portion of the fuselage which causes flow separation and drag. The particular four-digit airfoils specified were selected to give maximum lift and low drag for the thicknesses used on this design.

By M.A.Garl.M

Checked: Bayless

MODEL AIRPLANE REPORT NO

Aero Doc. Misc. #113
July 15, 1944

CENTER-OF-GRAVITY LIMITS

Aerodynamic C.G. limits have been estimated from wind-tunnel and flight-test data.

Definitions

Aft C.G. Limit

The aft C.G. limit is defined as that C.G. position (in per cent M.A.C.) at which the static longitudinal stability derivative, dCm/dC_L, equals -0.04 with flaps up and stick free. Limits are shown for two flight conditions:

a. Cruise power (approximately 50% normal rated power), level flight, $C_L = 0.7$ approx.

b. Normal rated power, climb, $C_L = 1.0$ approx.

The numerical value, $dCm/dC_L = -0.04$, has been found to indicate fairly reliably the minimum static longitudinal stability margin for satisfactory flight. The Stability and Control Research Section of the N.A.C.A. (Langley Field) has confirmed this value by correlation with free-flight wind-tunnel and full-scale flight tests.

Forward C.G. Limit

The forward C.G. limit is defined as that C.G. position (in per cent M.A.C.) at which full up-elevator deflection will trim the airplane at the maximum lift coefficient at landing, power off.

Hydrodynamic and ground handling C.G. limits are also shown.

BY *M. Gayfield*
CHECKED *Bailey*
APPROVED

MODEL AIRPLANE REPORT NO

Aero Doc. Misc. 113

C.G. LIMITS

% M.A.C.

plane	Hydro-dynamic or Ground Handling C.G. Limit	Aerodynamic Aft. C.G. limit Stick Free		Aerodynamic Forward C.G. Limit at Landing Power Off	Recommended C.G. Limits	
		Cruise Power Level Flight	Normal Rated Power Climb		Fwd.	Aft.
4J	34	30	28	23	23	28
24K	34	33	31	20	20	31
Y-2	34	33	31	20	20	31
el 39	33	33	31	20	20	31
32 ginal) Hori- al)	40	33	31	20	20	31
9	38	42	42	26	26	38
-5 t)	31 Hyd.	29	28	21 Aero. (24 Hydro.)	24	28
5A .)	34 Gnd.H. 31 Hyd.	29	28	21 Aero. (24 Hydro.)	24	28
-3 t)	34 Hyd.	32	30	23 Aero. (24 Hydro.)	24	30

: The basis for the above Aerodynamic aft. C.G. limits is shown on the following page.

BY

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Airplane	$\frac{dC_m}{dC_L}$ Power Off Stick Fixed C.G. 25% M.A.C.	Ref.	$\Delta\frac{dC_m}{dC_L}$ Cruise Power Level Flight	$\Delta\frac{dC_m}{dC_L}$ Normal Rated Power Climb	Ref.	$\Delta\frac{dC_m}{dC_L}$ Stick Free vs. Stick Fixed (Est.)	$\frac{dC_m}{dC_L}$ Stick Free C.G. 25% M.A.C. Cruise Power Level Flight	$\frac{dC_m}{dC_L}$ Stick Free C.G. 25% M.A.C. Normal Rated Power Climb	Aft C.G. Limit for dCm/dCL = -.04 Stick free Cruise Power Level Flight	Aft C.G. Limit for dCm/dCL = -.04 Stick free Normal Rated Power Climb
B-24J	-.16	ZT-32-012 (Galcit)	+.04	+.06	ZA-32-C86 (Flight Test)	+.03	-.09	-.07	30	28
XB-24K	-.22	ZT-32-012 (Galcit)	+.06	+.03	ZA-32-086 (Flt. Test)	+.04	-.12	-.10	33	31
PB4Y-2	-.21	ZT-100-004 (Galcit)	+.06	+.08	Galcit 445	+.03	-.12	-.10	33	31
Model 39	-.23	ZT-39-002 (Galcit)	+.08	+.10	Est. From Galcit 445	+.03	-.12	-.10	33	31
XB-32 (Orig. B-29 horiz. Tail)	-.17	ZT-33-007 (Galcit)	+.02	+.04	Est. From Galcit 287H	+.03	-.12	-.10	33	31
XC-99	-.27	Galcit 444	.00	.00	Est. From Part II NACA Power Tests, 1/14 Scale XB-36	+.06	-.21	-.21	42	42

BY.
CHECKED
APPROVED

10V

Airplane	$\frac{dC_m}{dC_L}$ Power Off Stick Fixed C.G. 25% M.A.C.	Ref.	$\Delta\frac{dC_m}{dC_L}$ Cruise Power Level Flight	$\Delta\frac{dC_m}{dC_L}$ Normal Rated Power Climb	Ref.	$\Delta\frac{dC_m}{dC_L}$ Stick Free vs. Stick Fixed (Est.)	$\frac{dC_m}{dC_L}$ Stick Free C.G. 25% MAC: Cruise Power Level Flight	$\frac{dC_m}{dC_L}$ Stick Free C.G. 25% MAC: Normal Rated Power Climb	Aft C.G. Limit for $dC_m/dC_L=-.04$ Stick Free: Cruise Power Level Flight	Aft C.G. Limit for $dC_m/dC_L=-.04$ Stick Free: Normal Rated Power Climb
PBY-5	-.12	Galcit 261	+.01	+.02	ZA-064 (Flt. Test)	+.02	-.09	-.08	29	28
PBY-5A	-.12	Galcit 261	+.01	+.02	ZA-064 (Flt. test)	+.02	-.09	-.08	29	28
PB2Y-3	-.18	Galcit 269	+.02	+.04	Est. From ZA-064	+.05	-.11	-.09	32	30

BY

CHECKED

Defendants' Exhibit B—(Continued)

Consolidated Vultee Aircraft Corporation
General Offices San Diego, California

Aero Memo #474
July 29, 1944.

Mr. T. P. Hall
Mr. M. A. Garbell

Wind Tunnel Tests of a 0.058 scale powered wind tunnel model of the thin wing Two-Engine Tailless Navy Design.

Enclosure (A) Plotted Data on Static Longitudinal Stability Flaps Up.

Wind tunnel tests of a 0.058 scale powered wind tunnel model of the thin wing Two-Engine Tailless Design (Aspect Ratio 12, Maximum wing root thickness 17%) have been in progress at Galcit since July 27, 1944. The purpose of the test is the determination of the general aerodynamic characteristics of this design with the revised wing. To date, power-off tests flaps up, including tuft photographs, have been completed.

Preliminary data indicates the same degree of static longitudinal and directional stability for this model as obtained on the previous 0.058 scale model of the tailless design incorporating the 22% thick wing (Enclosure (A)). Power tests are now in progress and the first data should be available during the first part of the coming week.

M. A. GARBELL.

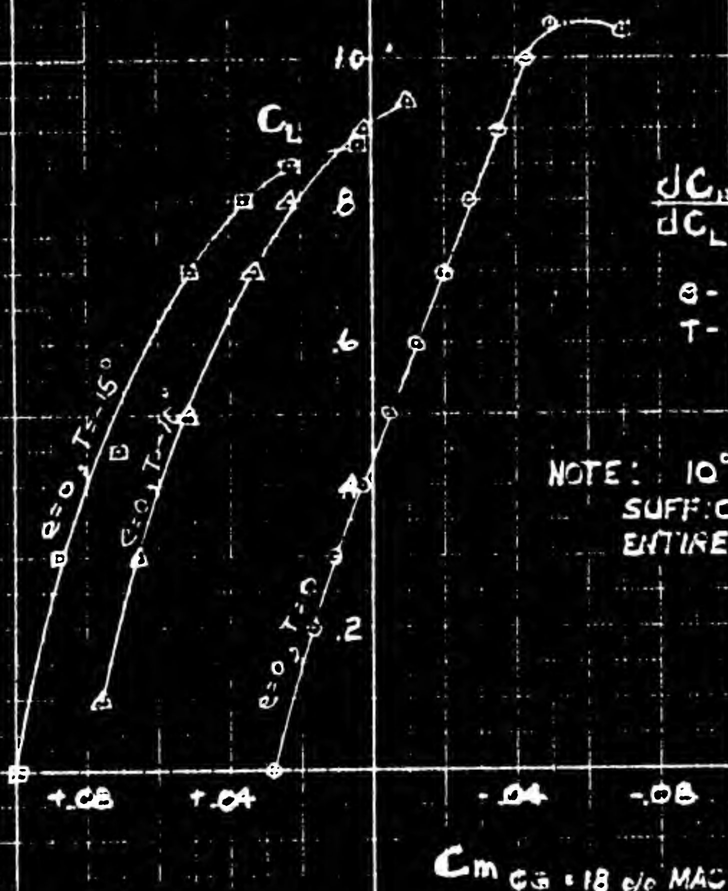
WES/lks
Aerodynamics Offc. #16
Dev. Engr. File

[In margin]: Filed, Hall.

PRELIMINARY DATA

THIN-WING TWO ENGINE TAILLESS NAVY DESIGN

COMPLETE MODEL
FLAPS UP - POWER OFF



$\frac{dC_m}{dC_L} = -0.075$

e - ELEVATOR
T - MID-SPAN TRIMMER

NOTE: 10° DEFLECTION OF TRIMMER SUFFICIENT TO TRIM OVER ENTIRE FLIGHT RANGE.

Defendants' Exhibit B—(Continued)

Consolidated Vultee Aircraft Corporation
General Offices San Diego, California

Ref.—Memo #2423
August 2, 1944

Mr. C. F. McCabe
Mr. T. P. Hall

Pressure Distribution—XB-32 Airplane.

(a) Aero Doc. #33-119 dated July 20, 1944. XB-32—Consideration of Pressure Distribution Measurements in Flight.

Enclosure (A) Copy of reference (a) to addresses only.

Mr. Sutton this date approved the referenced report and requested that we proceed with obtaining pressures as shown therein.

T. P. HALL,
Chief Development Engineer.

TPH/dmc

cc: R. L. Bayless
J. B. Jewell
C. B. Carroll
C. A. Phillips
D. K. Friday
Dev. Engr. Files

August 3, 1944 — To Garbell for work — not scheduled.

[In margin]: Garbell work to follow no schedule.

Defendants' Exhibit B—(Continued)

Consolidated Vultee Aircraft Corporation
San Diego Division

Page 1 of 6

Model 33 Airplane Report No. Aero Doc. #33-119

July 20, 1944

XB-32

Consideration of Pressure Distribution
Measurements in Flight

1. Wing

The possibility of determining the character of the airflow over the wing, in the region of the nacelles, by measurement of spanwise and chordwise pressure distributions have been studied. Available information indicates that pressure distribution data alone will not show up areas of flow separation. Figure 1 shows pressure and force data for a 66,2-414 airfoil section. The break in the lift curve at 6° angle of attack indicates trailing edge flow separation; however, the chordwise pressure distribution does not indicate this condition except possibly at 12° angle of attack where some loss in lift occurs over the trailing edge where the flow separation is very pronounced. The association of trailing edge flow separation with the break in the lift curve is based on previous tuft tests of the NACA 66 and 65 series sections.

As part of this study, a pattern for pressure orifices on the wing was laid out as shown by Figure 2. These orifices substantially cover the critical portion of the wing with a minimum num-

Defendants' Exhibit B—(Continued)

ber of pressure lines. An alternate method of obtaining pressure data is described in NACA report "The Belt Method for Measuring Pressure Distribution" dated February, 1943. This method requires the construction of a ¾″ wide pressure belt containing approximately 20, .040″ dia. copper tubes. The belt would be placed at about four spanwise stations on four different flights. This alternate method saves considerable work as compared to placing pressure orifices in the wing and also has the advantage of being readily adaptable to other areas if desired after analysis of the first preliminary data.

Pressure tests with this belt in conjunction with tuft observations should indicate the value of pressure data in determining the character of the flow. The tufts will show up the areas of flow separation or stall and it can be definitely determined if corresponding indications are present in the pressure distribution.

If the pressure data appear to be useful, a series of measurements may be made for several speeds varying from high speed to minimum cruise in level flight by 10 mph increments including climb with rated power. These data would be plotted as spanwise and chordwise distributions for study.

2. Fuselage

Pressure distribution measurements have already been made over the bomb bay doors of the XB-32 in flight as given in report ZA-33-023. Page 10 from this report is attached as a sample of the data obtained in these tests.

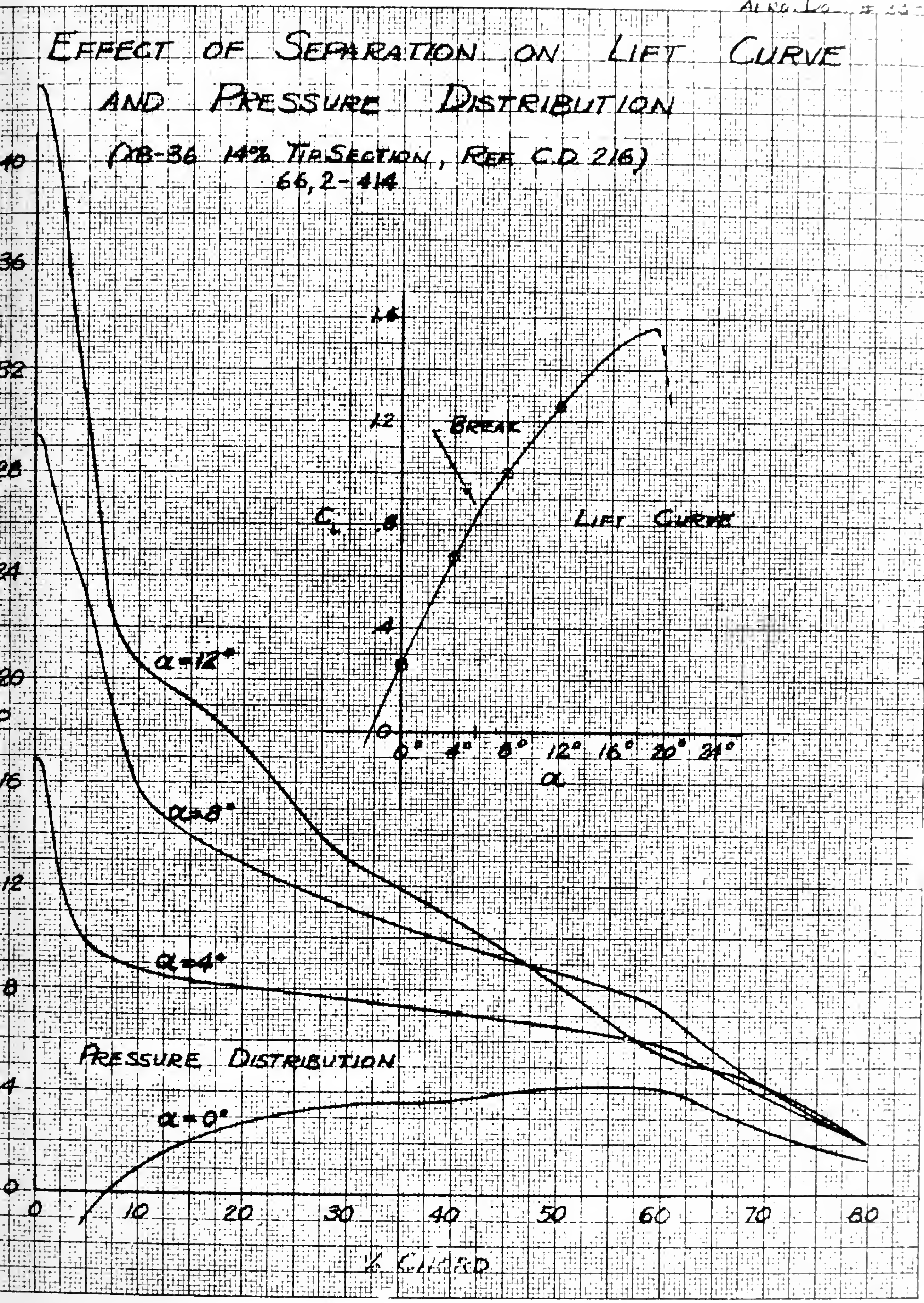
Defendants' Exhibit B—(Continued)

Other desirable pressure data on the fuselage may be obtained by installation of 16 pressure orifices around the pilots' enclosure, 6 orifices over the nose wheel door and 3 orifices on the fuselage side, as shown by figure 3. The data for the pilots' enclosure and the nose wheel door will be used to check structural analyses. The 3 orifices on the fuselage side will be used to investigate a position for a static orifice for the airspeed indicator. Pressure measurements may be recorded during other flight tests or a flight program similar to that proposed for the Model 39 in report ZA-39-021 may be used.

By /s/ C. L. BLAKE.

Checked /s/ BAYLESS.

Approve ....


EFFECT OF SEPARATION ON LIFT CURVE
AND PRESSURE DISTRIBUTION
(XB-36 14% TIP SECTION, REF C.D. 216)
66,2-414
BREAK
LIFT CURVE
C_L
α
α=12°
α=8°
α=4°
α=0°
PRESSURE DISTRIBUTION
% CHORD

BLAKE
7/18/44

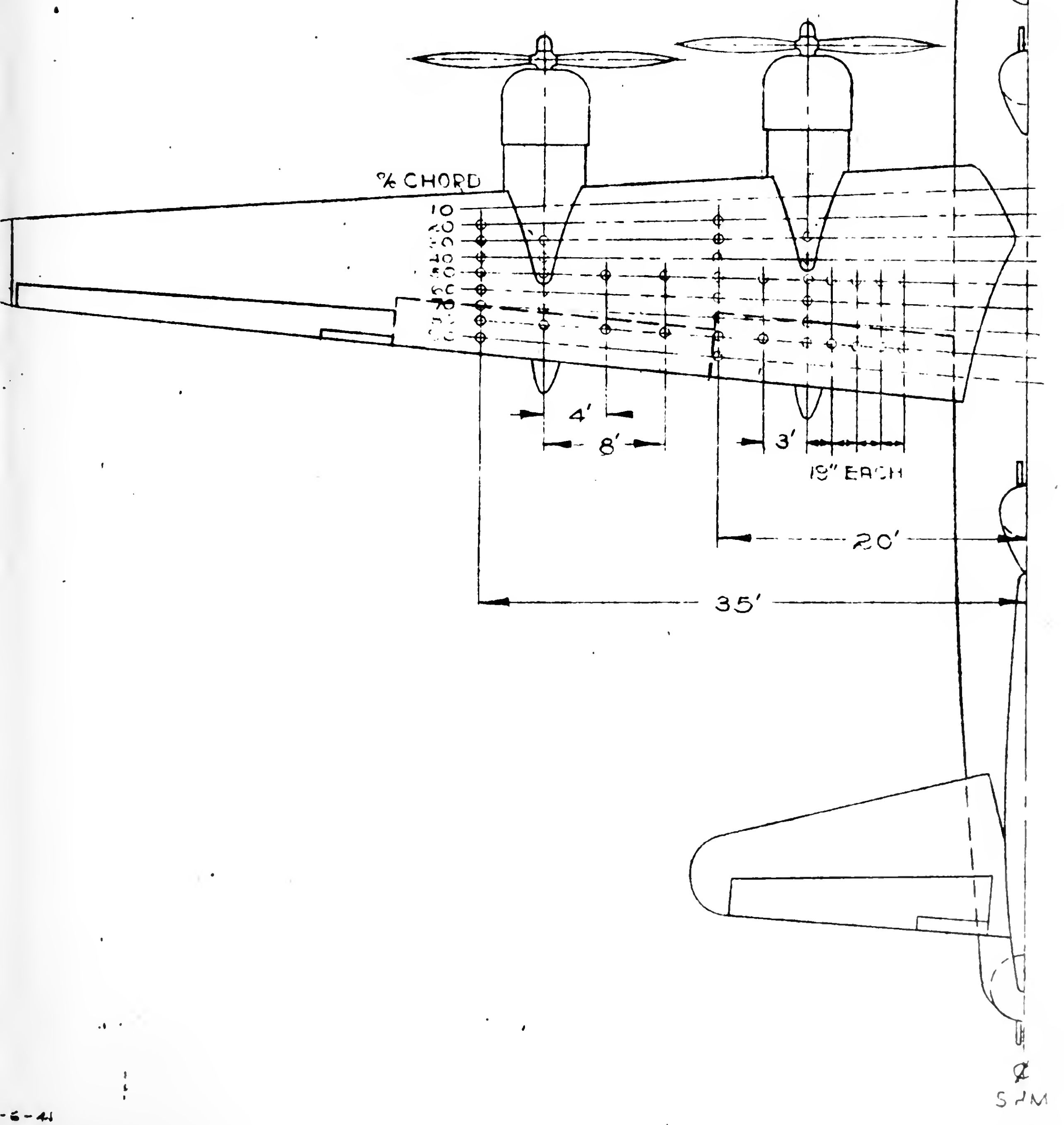
FIG. 2

CONSOLIDATED VULTEE AIRCRAFT CORP.
SAN DIEGO

Aero Doc. #33-119
July 14, 1944

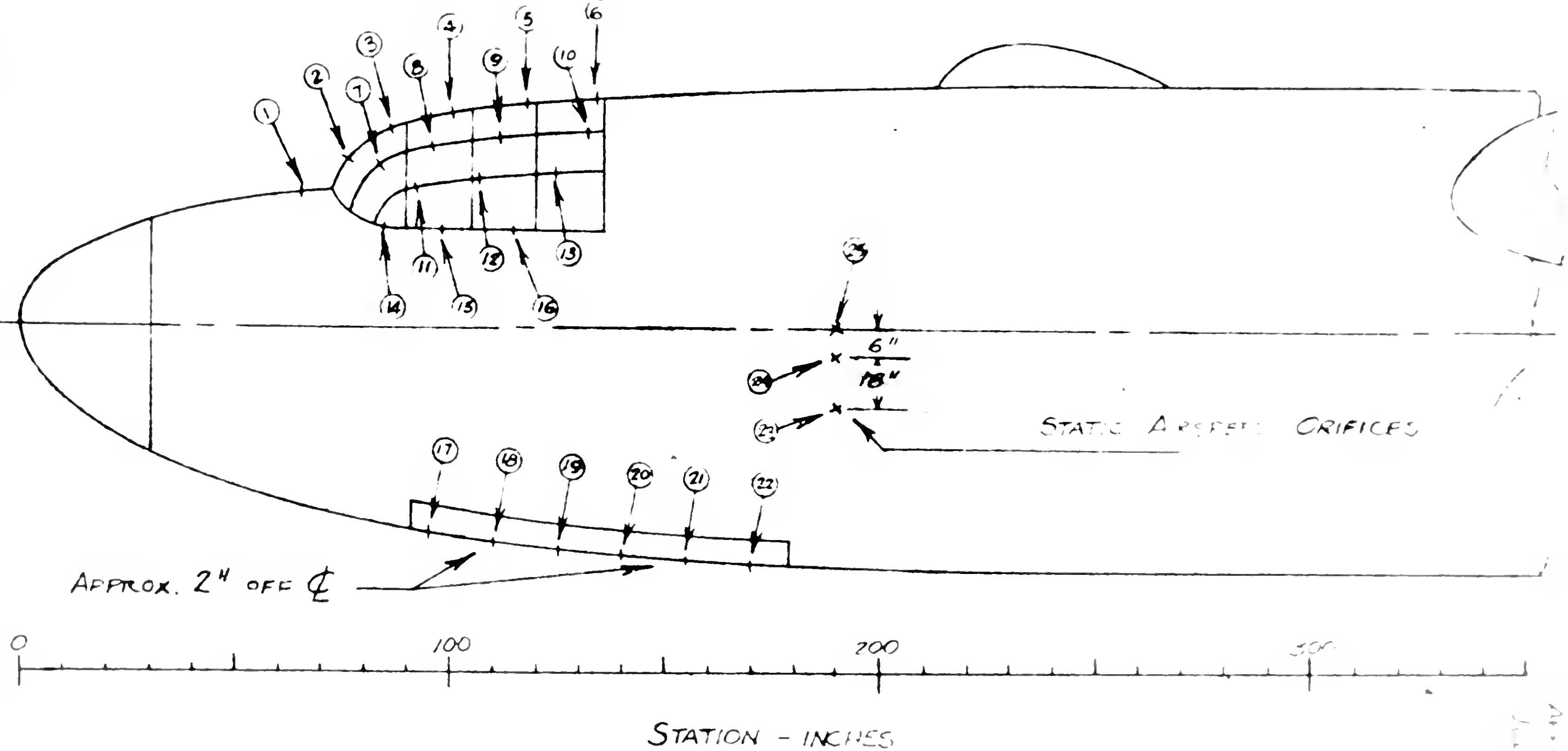
PROPOSED LOCATION PRESSURE ORIFICES
FOR XB-32 FLIGHT TESTS

(UPPER SURFACE ONLY)



-6-41

PROPOSED PRESSURE TUBE LOCATIONS FOR XB-32 FUSELAGE



DRAWN	CLE	7/20/41	PART NUMBER
APPROVED			
APPROVED			

CONSOLIDATED AIRCRAFT CORPORATION
LINDBERGH FIELD • SAN DIEGO, CALIFORNIA

743

Defendants' Exhibit B—(Continued)

Intra-Company Correspondence
Consolidated Vultee Aircraft Corporation
General Offices San Diego, California

Aero Memo #481
Date 6 August 1944

From: Mr. T. P. Hall

Subject: Mr. M. A. Garbell

Reference: Preliminary Comments on Wind-Tunnel Tests of 0.058 Scale Powered Model of Two-Engine Tailless Design (Aspect Ratio 12, Thin Wing)

Enclosure: (A) Summary Table of Aerodynamic Characteristics

A .058 scale model of the two-engine tailless design was tested with and without running propellers and with no airflow through the nacelles. This new, higher-speed version of the design compares favorably with the thick-wing version tested at Galcit and M.I.T. (ref. CVAC Report ZT-029 and Appendices). As no tunnel tares were made for these tests the drag values obtained are not reliable. The new thin aft extendible surface is inadequate because the modified aft-surfac airfoil did not equal the high-lift characteristics of the one previously tested.

The model should be reworked before it is sent to Moffett Field. Suggested construction changes are being analysed and the final recommendations will be given to the model shop as soon as possible. The model should be ready to go to Moffett Field at the end of August.

Preliminary Galcit plots of the tests should be

Defendants' Exhibit B—(Continued)

available at San Diego by the 10th or 11th of August. The data on the following summary table of the test results were obtained during the test and are unchecked. A report will be written within a week after the Galcit data reach San Diego.

[In margin]: File misc. Pl. don't return to T. Hall.

/s/ MAG.
M. A. GARBELL.

MR:hes
cc: Aerodynamics
Devn. Engr. File

[In margin]: Memo.

[In margin]: Memo tail camber status.

HAS from TPH.

Enclosure (A)
Aero Memo #481

CONSOLIDATED VULTEE AIRCRAFT CORPORATION
GENERAL OFFICES . . SAN DIEGO, CALIFORNIA

SUMMARY TABLE OF AERODYNAMIC CHARACTERISTICS

		Flaps 0°	Flaps 40°
aximum Lift Coefficient max (ΔC_L = 0.6 added for extrapolation to full scale)		1.6	2.3 Same
atic Longitudinal Stability rivative dC_m/dC_L	Props off	-.074	--
	Windmilling Power	--	-.130
	Cruise Power (.40 NRP)	-.040 was -.05	--
G. @ 18% Flaps up	Normal Power	-.030	--
G. @ 19% Flaps down	Take-off Power	--	Data not yet Available
atic Directional Stability rivative, $dC_n/d\psi^\circ$, (props off)		-.0006 Same	Data not yet Available
d-span trimmer effectiveness $m/d\delta t^\circ$	Props off	-.0055	--
	Cruise Power (.40 NRP)	-.0055	---
	Normal Power	Data not yet Available	--
evator effectiveness, $m/d\delta e^\circ$	Props off	Data not yet Available	--
	Props Windmilling	--	Data not yet Available
	Cruise Power (.40 NRP)	-.0016	--
	Normal Power	--	--
	Take-off Power	--	Data not yet Available
an Efficiency, "e", between $_L$ = 0.3 & C_L = 0.6 (high speed & cruise)		.925 better	--
leron Control Criterion b/2V (obtained with no change in trim) ruise power (.40 NRP), with full control eflection ± 20° aileron and spoiler		.08 Same	--

MODEL 32 AIRPLANE REPORT NO Aero Doc. /32-109 July 18, 1944

Revised August 18, 1944

B-24D

TAIL LOADS IN LEVEL, UNACCELERATED FLIGHT

Figure 1 shows the tail loads for the B-24D in level unolerated flight between high speed and the speed for maximum range. loads are positive (i.e. up) except for high speed at very low ght. The data are shown for 25,000 ft.

Figure 2A shows the tail loads in terms of (tail load/q) tted vs. C_L. Figure 2B shows the pitching moment vs. C_L with tail . The tail load is computed from the unbalanced pitching moment follows:

$$\text{Tail load (lbs.)} = \frac{C_m S C q}{l_t}$$

Where S = wing area = 1,048 sq. ft.

C = MAC = 10.3 ft.

l_t = tail length = 36.5 ft.

q = dynamic pressure

= .00256 V_i^2

Where V_i = true indicated airspeed in mph.

Figure 3 shows C.G. data for the B-24D. A value of 32% assumed to be representative for the computation of tail loads.

Figure 4 shows C_L vs. true indicated airspeed for several ghts. These data are for reference only.

STATIC LONGITUDINAL STABILITY IN LEVEL, UNACCELERATED FLIGHT

The actual value of the tail-loads has no effect on the gitudinal stability of the airplane. The important element is the l-load slope. With increasing angle of attack the tail-loads on B-24 airplane increase in a positive (up) sense, thus producing ater diving moments; this variation is stable as shown in Fig. 2 B.

BY M.A. Gaskill

CHECKED B. Naylor

APPROVED

July 13, 1944
MODEL 32 AIRPLANE REPORT NO. Aero Doc. #32-109

STRUCTURAL CRITERIA FOR TAIL LOADS

The B-24 tail surfaces are designed for the loads arising in four principal flight conditions as follows:

1. Balancing loads at the four corners of the V-g diagram, i.e. the design load factor at

   (a) High angle of attack (up tail load)
   (b) Low angle of attack (up tail load)
   (c) Inverted flight, high angle of attack (down tail load)
   (d) Inverted flight, low angle of attack (down tail load)

2. High speed, one -"g" flight with a 30 ft/sec. up or down gust. (Tail load up or down depending on direction of gust.)

3. Pullout (tail load first down and then up).

4. Placard speed with flaps down and 30 ft/sec. gust (tail load down).

The B-24 tail is designed, by the critical up and down tail loads. The tail loads for other designs may be in the opposite sense in some cases depending on the design conditions.

BY

CHECKED

APPROVED

CONSOLIDATED VULTEE AIRCRAFT CORPORATION
GENERAL OFFICES . . . SAN DIEGO, CALIFORNIA

Aero Memo #537
October 10, 1944

Mr. T. P. Hall

Mr. C. L. Blake

Current Wind Tunnel Tests on the 2-Engine Executive 1/8-Scale Preliminary Power-Off Model at Galcit.

Enclosure: (A) Aero Doc. Misc. #138 dated October 10, 1944.

The attached sheets show a summary of the tests to be conducted and sketches of the various fillets to be tried in selecting the basic airplane configuration.

C. L. Blake

JP:jm
cc: Dev.Engr.File
Aero [illegible]

The wing referred to as the "44" wing is actually a [illegible] following [illegible] airfoils [illegible]

[illegible] 44

Defendants' Exhibit B—(Continued)

Consolidated Vultee Aircraft Corporation
General Offices San Diego, California

Aero Doc. Misc. #138
October 10, 1944

Test Outline 2-Engine Executive
1/8 Scale Wind Tunnel Model

1. Strut tares and flow inclination determination——

a. Wing alone, NACA 44 and 63 section wings.

b. Complete model using each wing.

2. Wing alone study: Selection of either the NACA 44 or 63 section wing.

a. Tuft studies—stall hysteresis analysis.

3. Model build-up drag analysis.

4. Flow investigation near Wing - fuselage, Wing-Nacelle and Fuselage-tail intersections. Tests of necessary fillets to improve flow conditions will be made.

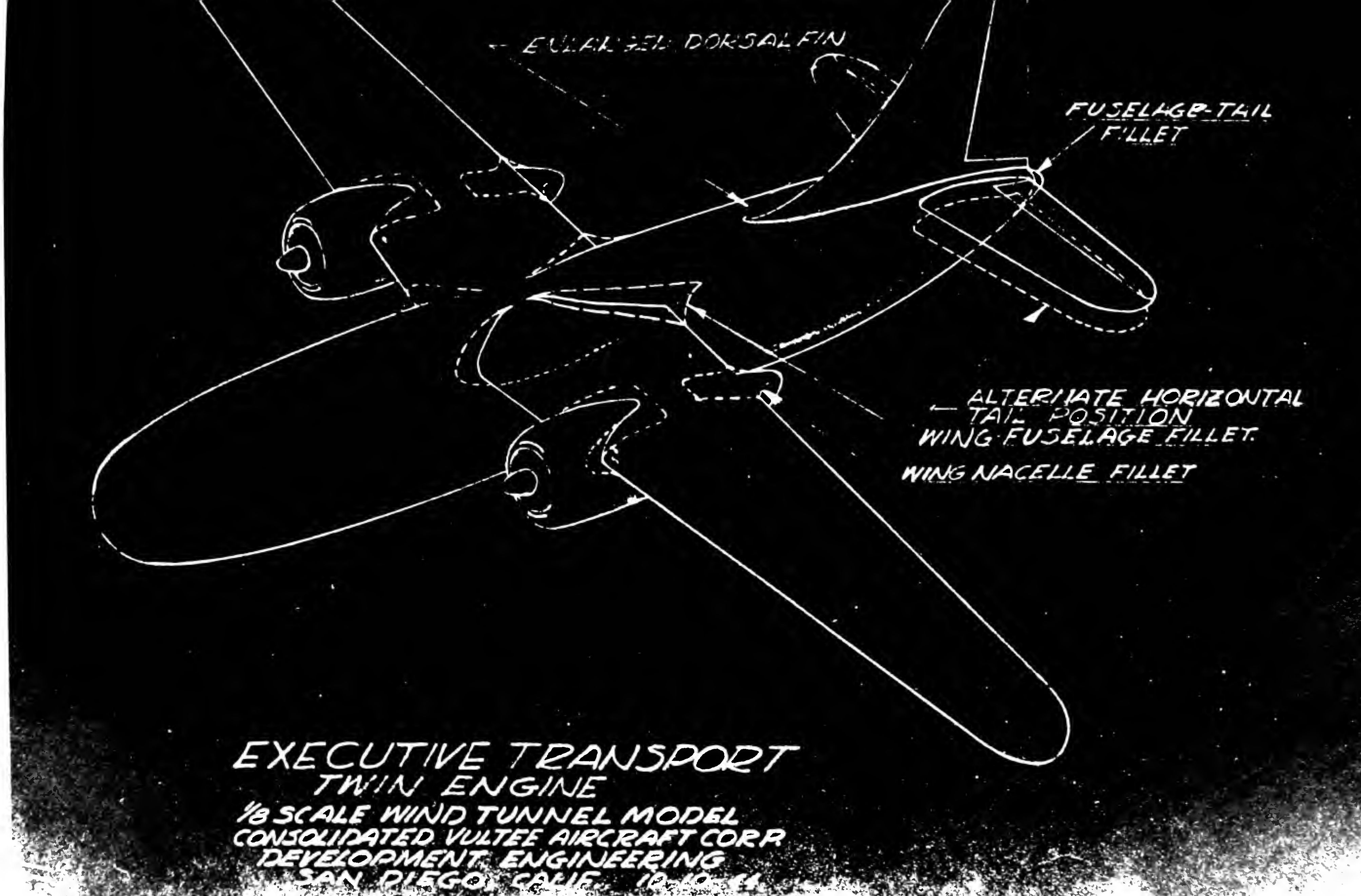
5. Total head survey, with flaps extended and retracted, to locate best tail position.

6. Longitudinal stability and control, elevator effectiveness, flaps zero and fully deflected for final selected wing and complete model including fillets (and tail off).

7. Directional stability and control, rudder effectiveness for complete final configuration flaps 0° (and tail off).

8. Stabilizer effectiveness using both wings.

9. Test of a larger dorsal fin.


ENLARGED DORSAL FIN
FUSELAGE-TAIL
FILLET
ALTERNATE HORIZONTAL
TAIL POSITION
WING-FUSELAGE FILLET.
WING NACELLE FILLET
EXECUTIVE TRANSPORT
TWIN ENGINE
1/8 SCALE WIND TUNNEL MODEL
CONSOLIDATED VULTEE AIRCRAFT CORP.
DEVELOPMENT ENGINEERING
SAN DIEGO, CALIF.

Defendants' Exhibit B—(Continued)

Consolidated Vultee Aircraft Corporation
San Diego, California

Page 1 of 4

Aero Doc. Misc. #142
November 3, 1944

Model Airplane Report No.

Comments on Stinson Report No. 1551
Series II Wind Tunnel Test of 1/5 Scale

Powered Model Single Engine Pusher Design
(Reference: MIT Report #651)
October 14, 1944

A study of the subject report indicates that, despite the installation of the large leading-edge slot, the stalling characteristics of the airplane remain unsatisfactory, especially with flaps deflected. This is particularly borne out by the data plotted in figure 14 of the subject report (figure A attached to this Aero Document), which shows that even a small deflection of the elevator causes a breakdown of the airflow about the wing and a loss of lift of $\Delta CL = -0.4$. A typical satisfactory airplane is shown, for comparison, in figure B. The airflow conditions with flaps retracted are also unsatisfactory as indicated by the following test material:

1) Figure 11 (page 19)—Most curves show objectionable discontinuities in the static longitudinal stability slopes.

2) Figure 13 (page 21)—The sharp variations of the rolling and yawing moments, as well as side forces, indicate asymmetric local stall phenomena which would contribute to make the stall of the airplane vicious and diffi-

Defendants' Exhibit B—(Continued)

cult to control. A comparison with the characteristics of the original model with the leading-edge fillet, shown in figure C, indicates a deterioration in this respect.

3) Photograph on page 49—Despite the installation of the large slot, a distinct cross flow appears between the fuselage and the tail booms, indicating the existence of turbulent separation at the wing-fuselage intersection.

Conclusion:

The new model with the slotted inboard panel shows no substantial and consistent improvement over the optimum previous model configuration with the leading-edge fillet which was not considered a satisfactory basis for further design and construction work. The drag difference of .0010 between slot and leading-edge fillet is not representative of the actual drag difference between the two modifications, because of the high surface drag of the leading-edge fillet which consisted of a basic wood structure and a large amount of plasticene.

By /s/ M. A. GARBELL.

Defendants' Exhibit B—(Continued)

Consolidated Vultee Aircraft Corporation
San Diego, California

Page 1 of 2
Aero. Doc. # TL-105

December 26, 1944

Model........ Airplane........ Report No.......

Preliminary Proposal for a Scale Model of the Two-Engine Tailless Airplane

The construction of a scale model of the two-engine tailless airplane, large enough to accommodate a pilot as well as a radio control and recorder, is proposed to obtain additional information on the stability and control characteristics of the tailless design at a scale which approximates more fully that of the actual airplane. The model should be tested in free flight and as a static wind-tunnel model in the "full-scale 80′ x 40′ tunnel" at Moffett Field.

It is proposed to use existing Navy radio equipment if radio controlled flight testing is desired.

Model Characteristics
0.4 Scale—No Power

This model, geometrically similar to the full-scale design, would yield valuable information on stall, stability and control characteristics at a high Reynolds number as well as the "feel" of the airplane,

Defendants' Exhibit B—(Continued)

and would permit the investigation of the most desirable path and hinge moments of flaps and control surfaces. It would also serve to study and develop additional means of obtaining greater directional stability at a minimum cost and risk.

General Data

Scale	0.4
Span	58.8 ft.
Wing Area	288 sq. ft.
Fuselage Diameter	41.6 in.
Gross Weight	1440 lb.
Type of Construction	All wood

Consolidated Vultee Aircraft Corporation
San Diego, California

Page 2 of 2

Aero. Doc. #TL-105
December 26, 1944

Model Airplane Report No.

Estimate of Man Hours

Item	Man Hours
Structural layout and design	
4 men for 4 weeks	800
Structural analysis	
1 man for 4 weeks	200
Shop time (mostly in model shop)	
12 men for 10 weeks	6,000
Total	7,000

Defendants' Exhibit B—(Continued)

This number of man hours is equivalent to that of two power-off wind-tunnel models of much smaller scale.

Provisions should be made to incorporate fittings for the balance of the Moffet Field "full-scale" tunnel.

Any airplane of the 100-150 HP class will be sufficient for towing this model.

Additional Consideration

If the power off tests give reliable and encouraging data, it is suggested that the tests be extended to include a dynamically similar 0.4 scale model powered with two Lycoming 0-290 engines (130 BPH each which will simulate full take off power). This model would be suitable for complete wind tunnel and flight tests and for presentation to the trial boards of potential customers.

By /s/ M. A. GARBELL.

NO 184-R

CONSOLIDATED VULTEE AIRCRAFT CORPORATION
SAN DIEGO, CALIFORNIA

MODEL AIRPLANE REPORT NO Aero. Doc. #TL-106

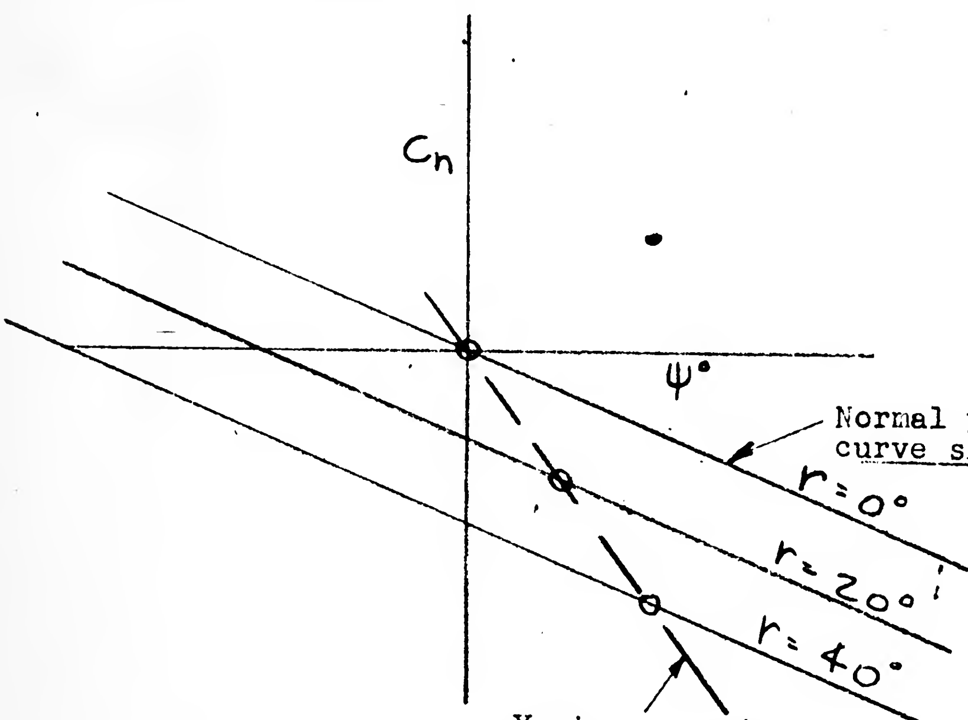
December 26, 1944

TWO ENGINE TAILLESS

Study of Means to Increase

Directional Stability

Theoretical studies show that increased static directional stability may be obtained by use of a horn balance or a large leading edge balance which will produce overbalance of the rudder in yaw. The resulting anti-trail of the rudder will increase the static directional stability as shown below.



The inherent fault of the above system is the hunting characteristic for which no satisfactory corrective means has been determined.

The static directional stability may be increased also by enlarging the wing tip surfaces; however, it does not appear feasible to increase the area of these surfaces beyond 15% of the wing area (Note: Present two-engine tailless design has 12% surfaces). The resulting $\frac{dCn}{d\psi°}$ would be only -.0008 with 15% surfaces as compared to the present -.0006. This value is about one-half of the $\frac{dCn}{d\psi°}$ for the PB4Y-2 and B-32.

BY

CHECKED

APPROVED

Consolidated Vultee Aircraft Corporation
San Diego, California

Page 2 of 2

Aero. Doc. #TL-106
December 26, 1944

Model .... Airplane .... Report No.

Two Engine Tailless

An effective increase in the directional stability may be obtained by an increase in the directional damping of the airplane. This may be accomplished by connecting the rudder control permanently to the yawing velocity channel of the automatic pilot. The rudder will automatically counteract a tendency to yaw by building up a restoring moment at a rate equal to the magnitude of the disturbance. The resulting effect will be to increase the directional stability in the same manner as would be obtained with greater fin area. It has been calculated that this arrangement can be adjusted simply to give a directional stability equivalent or possibly superior to a $\frac{dCn}{d} = -.0018$ which is representative of current conventional airplane design. This arrangement may be tried on a twin tail B-24 to determine the degree to which the directional stability can be improved.

Defendants' Exhibit B—(Continued)

Consolidated Vultee Aircraft Corporation
San Diego, California

Page 1 of 15

Model .... Airplane .... Report No.

Aero Doc. # Misc. 192
May 10, 1945

Report on
Conferences at Ames Aeronautical Laboratory
Moffett Field, California

May 4, 5, and 7, 1945

A series of conferences were held at the Ames Aeronautical Laboratory, between representatives of NACA and CVAC, on 4, 5 and 7 May 1945, to discuss the forthcoming tests of the XB-46 design in the Moffett Field wind tunnels and to exchange opinions and ideas on certain aerodynamic high-speed problems relating to this design.

NACA Representatives

D. H. Wood
C. W. Frick, 7′ x 10′
R. Jackson, 7′ x 10′
M. J. Hood, 16′
W. T. Hamilton, 16′
J. Allen
(Others were met in informal conversations)

CVAC Representatives

M. A. Garbell, Development
G. L. Shue, Aerodynamics, San Diego

By /s/ M. A. GARBELL.

SUMMARY OF SUBJECTS DISCUSSED

I. 7' x 10' Wind-tunnel Test of 0.075-Scale power-on Model.

II. 7' x 10' Wind-tunnel Test of 0.3-Scale Semi-Span Horizontal Tail.

III. 16' Wind-tunnel High-speed Test of 0.09-Scale Power-off Model.

IV. Wing Airfoils.

V. Tail Airfoils.

VI. Effects of Jets on Longitudinal Stability.

VII. Interference between Jets.

III. Nacelles and ducts.

IX. Flush scoops.

X. Effect of Nacelles on Span-Load Distribution.

XI. Lateral Control.

XII. Dive Recovery Devices.

III. Canopy.

XIV. Photographs of Compressibility Shock Fronts.

XV. Effect of Wing and Tail Shock Fronts on Control Forces.

VI. Airflow through the Bomb Bay at High Speeds.

II. Determination of the Critical Mach number of three-dimensional Bodies.

II. Availability of NACA Memorandum Reports for AAF and BuAer.

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I. 7' x 10' Wind-Tunnel Test of 0.075-Scale Power-on Model.

1. The wind tunnel will be available for testing the XB-46 model beginning 21 May 1945.

2. NACA expects to start the test 2 weeks after arrival of the model at Ames Aeronautical Laboratory (Estimated 15 May 1945).

3. The test period is expected to last for 3-4 weeks.

4. Model drawings should be sent to NACA at once, for inspection and structural check.

5. Jet unit should be sent to NACA at once, for bench test alone and in conjunction with rear-strut attachments.

6. Use data at various $\frac{\check{v}}{V}j$ ratios to obtain "power-off" and "idling power".

7. Comments on CVAC test program (Aero Doc. #109-114, Revised May 1, 1945):

   a. Ref. I, A (Purpose): A new AAF Spec. C-1815a is being distributed to replace Spec. C-1815.

   b. Ref. I, D (Tests), par. 1: Tab effectiveness should not be included on this small model.

   c. Ref. I, D (Tests), par. 5. Omit this test, use cross-plots of hinge-moments instead.

8. NACA is fully equipped to take tuft movies if necessary.

9. Small lift and pitching moment tares with jet-power-on are anticipated (approx. 2 lb. ΔL and 0.5 ft. lb. ΔM).

10. Perfect alignment of all control-surface hinges is an absolute prepequisite to the attainment of good hinge-moment data.

11. NACA recommends that the nacelles be painted in the customary manner despite the fairly high temperatures of the primary jet air.

12. It was agreed that the tests be commenced with the ground board runs, power on and power off, in order to determine the adequacy of the horizontal tail and the behavior of the jet close to the ground. Great importance is attributed to this phase of the test, the first of its kind ever performed on a multi-jet design.

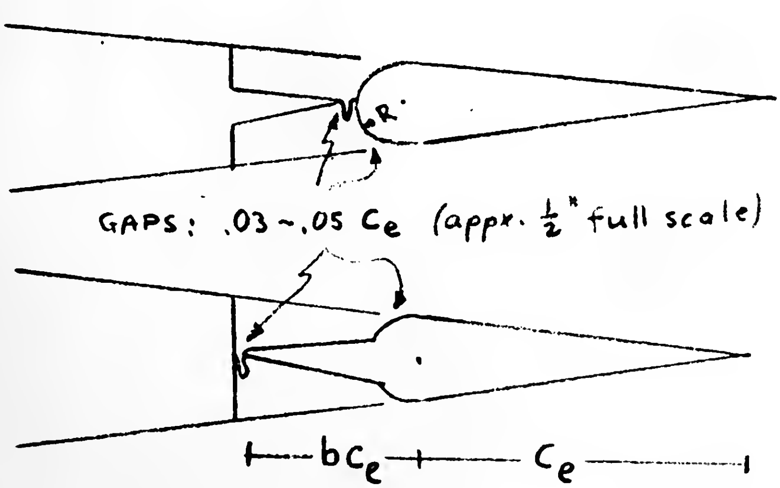
7' x 10' Wind-Tunnel Test of 0.3-Scale Semi-span Horizontal Tail.

1. This test is expected to start approximately 4 weeks after the start of the 0.075-scale model test, and to last approximately 2 weeks.

2. Drawings should be completed and sent to the NACA as soon as possible. Actual construction of the model, however, should await the results of the ground-board test of the three-dimensional model.

3. The model must have steel spars in both the stabilizer and elevator and must be designed for q = 80 lb/ sq.ft. (ultimate load factor 5).

4. Two alternate internally sealed nose balances (see sketch below) shall be tested to provide means of calculating the characteristics of any intermediate balance.



Nose balance scale consisting of dental dam as enclosed here will be furnished and installed by NACA, but CVAC must

provide for means of installation. The seal gap should be very small to avoid non-linear jumps in the hinge-moment curve.

Balance cell pressures shall be taken at four span-wise stations.

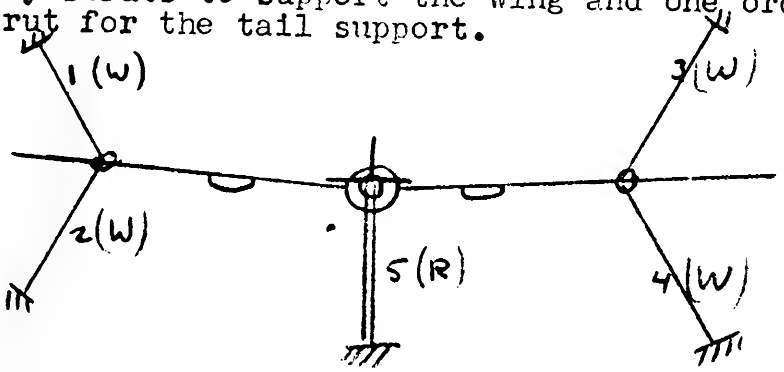
5. External pressure tubes shall be taken on both sides of the airfoil and shall extend as close as possible to the trailing edge.

6. Control tabs shall be aerodynamically balanced. The NACA prefers to install their own hinge moment strain gages; CVAC, however, is expected to install the electric leads from the tab strain gage location to the elevator torque tube and along the torque tube center line through the wind-tunnel wall.

I. 16' Wind-tunnel high-speed test of 0.09 Scale power-off Model

1. At present, the 16' high-speed tunnel is scheduled for high-priority tests through 15 September 1945.

2. The new suspension system consists of four tension-only struts to support the wing and one ordinary strut for the tail support.



This new system eliminates local choking at moderately high Mach numbers.

3. The wing cannot be supported without faired bumps at the trunnions.

4. Yawing and rolling moments are not accurately measured with this suspension system. High-speed vertical tail characteristics must be determined from pressure distribution data in the Co-op tunnel.

5. The tail trunnion must be accessible both up and down for tare runs with the vertical tail off.

6. The principal problems arising in all tests in the 16' high-speed tunnel are:

   a. buffeting and shaking of the models, especially at high speeds.

   b. a very large temperature range, affecting the strain gage readings.

   c. the ample range of q's.

7. Specifications for strain gages:

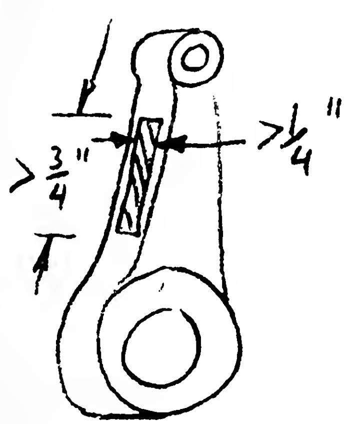
   working stresses (with ultimate load factor 5)

	Steel	Al. Alloy
Bending gages	S_f = 32,000 lb/sq.in	S_f = 12,000 lb/sq.in
Torsion Gages	S_f = 17,000 lb/sq.in	S_f = 7,000 lb/sq.in

   All strain gages should be supplied at least in duplicate

8. Minimum size of strain gages:

   a. Bending gages



   Approximately constant-stress beam (with straight sides). Strain gages on both sides forming the opposite branches of the bridge in order to minimize temperature effects.

   b. Torsion gages:

   1/8" diameter   1/2" length

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9. Angle indicator drives must have their own link systems not attached to the strain gages. Temperature effects should be eliminated by connecting all four branches, or by using selsyn drives (for example, Kollsman #845-01).

10. All hinges must be perfectly aligned and must be built very sturdily to resist the considerable shaking of all control surfaces at large deflections. Control surfaces must be mass balanced.

11. Remote-control drives and position indicators should be considered to replace manual positioning. The desirability of the various remote controls is expressed by the following order of priority:

    1. elevator
    2. rudder
    3. ailerons (if possible)

    Special note on elevator and rudder: One actuating motor (for example, a Learavia actuator) located far ahead inthe fuselage (perhaps in the bomb bay) to avoid interference with the tail trunnion, may alternately drive the elevator and the rudder merely by switching the driving links.

    The two elevator halves may be controlled separately; hinge moments may then be measured on one semi elevator, while the other semi elevator is used for pressure distribution measurements.

    All actuating mechanisms must be very rigid.

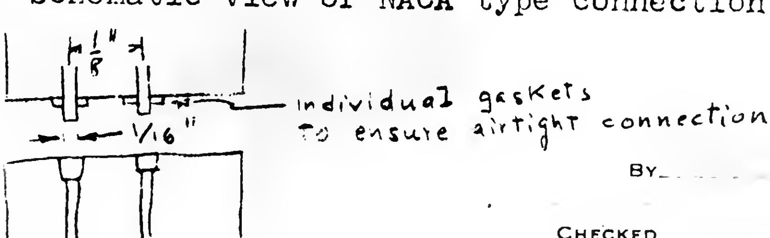
12. Notes on pressure distribution measurements.

    a. Approximately 150 tubes can be easily accommodated simultaneously (more if necessary).

    b. Four wing pressure distribution stations should suffice.

    c. All copper tubing must be annealed to avoid cracking.

    d. Schematic view of NACA type connection plugs:



13. Note on induction air flow through nacelles:

No powered blower is needed for $\frac{v}{V}$ up to approximately 0.8. Baffle plates should be provided for lower inlet-velocity ratios.

14. Notes on CVAC preliminary schedule:

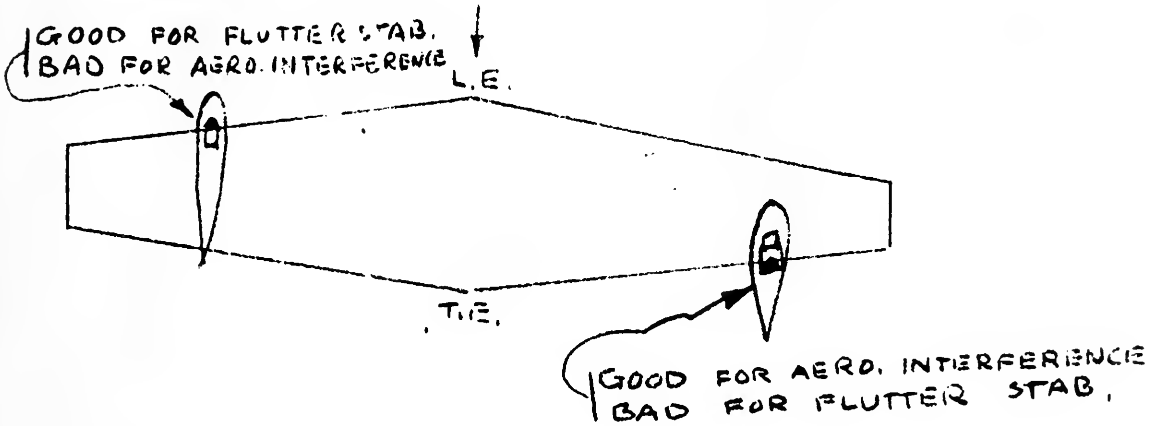
a. Three stabilizer settings are considered to be fully adequate.

b. Gear hinge moments should be measured with the landing gear extended (it is also suggested that an outside pressure distribution on the tires be run, if the landing gear is to be extended at high speed).

15. The stress analysis of the model should be based on the following design conditions:

q = 800 lb/sq.ft.
M in excess of 0.85
n_{ult} = 5

This will necessitate an all steel wing (or similar strong material). A complete flutter analysis will be required by NACA. The two extreme solutions for the design of the wing support trunnion are shown in the following sketch:



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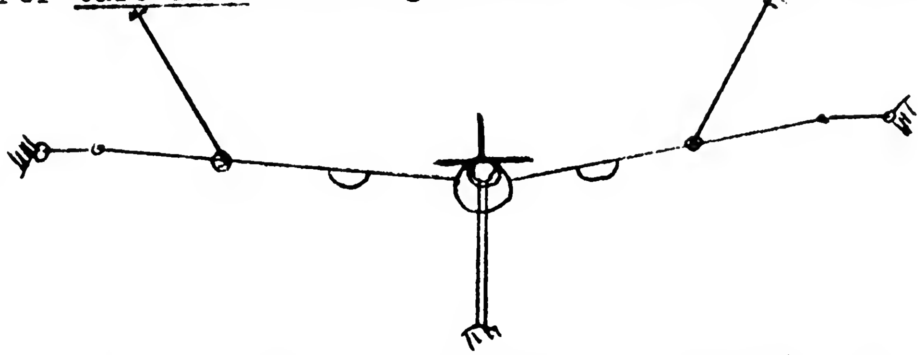
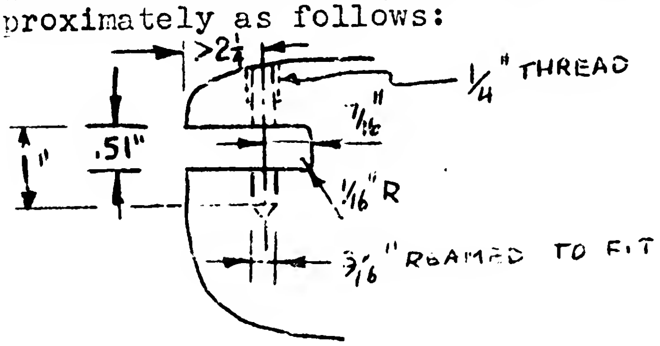
16. For tare runs the wing must be supported as follows:



The two thin tie-rods intended to restrain any lateral motion of the model require fittings in the wing approximately as follows:



17. The NACA is equipped to take tuft movies. The interference of the bumps at the strut-wing intersection, however, will reduce the significance of any tuft studies greatly.

IV. Wing Airfoils

1. No serious objection against the "straight-sided" fairing of the XB-46 wing airfoils was voiced by any NACA representative.

2. No information on the physical laws governing the development of two compressibility shock fronts on an airfoil in the deflected flap (aileron) is available.

3. NACA representatives have no knowledge on optimum flap gaps for thin 65 sections. A new Memo Report for Bumer on 66-216, a = 0.6, with flaps, by Holtzclaw, shows an optimum chordwise position of the flap 2% ahead of the physical wing trailing

edge (same as our Galcit test) but indicates larger and less critical gap values. At RN = 5×10^6 the BuAer test shows the same C_{Lmax} with flaps deflected for the following two configurations:

$\Delta C_{D_P} = .0015$

$\Delta C_{D_P} = 0$

FLAPS RETRACTED

V. <u>Tail airfoils</u>

1. The following fairly recent report, not available at CVAC, should yield the needed information:

   Henry Jessen: The Effect of Various Horizontal Tails upon the High-Speed Longitudinal Control of the P-51B Airplane from Wind Tunnel Tests. NACA CMR for AAF, 24 June 1944.

   Also request preliminary data on a P-47 test with spoilers on the horizontal stabilizer from the Army.

2. It was agreed that a 64_1 - 010 or 64_1 - 011 should replace the 66_1 - 010 <u>airfoils</u> on the XB-46 <u>tail</u> surfaces, in order to minimize the adverse compressibility effects due to control surface deflections and to reduce the sensitivity of the tail surface airfoils to surface roughness. (It may be necessary to maintain higher-than-static pressures inside the movable surfaces in order to minimize hinge-moment troubles due to skin deflection. M.A.G.)

3. No data on $\frac{\Delta V}{V}$ due to control surface deflection are available for the 64 airfoils. Some information may be gleaned from J. Allen's TR 637. Allen also suggests that these increments may be estimated from the increments measured on the 66_1 - 010 airfoils by using the following expression:

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MODEL .... AIRPLANE   REPORT NO Aero Doc #Misc 192

$$\left(\frac{\Delta V\delta}{V}\right)_{64} = \left(\frac{\Delta V\delta}{V}\right)_{66} \frac{\left(\frac{V}{V_0}\right)_{64}}{\left(\frac{V}{V_0}\right)_{66}}$$

This expression neglects the change due to difference in airfoil thickness at the control surface hinge line.

4. The 64 section may require more nose balance than the 66 section because of its smaller T.E. Angle.

5. NACA recommends ribbed construction on balance shroud with balance nose notches to clear, thereby permitting greater deflections with large balance noses.

6. NACA recommends tests with 0 and 40% balance, with pressures taken in the balance cell.

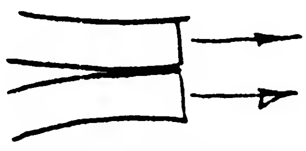
I. Effect of Jets on Longitudinal Stability

1. NACA recommends use of the method by Squire and Trouncer presented intheir report on "Round Jets in a General Stream".

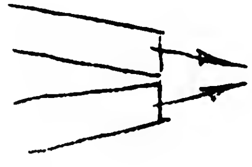
2. Our own estimate of $\Delta \frac{d\ C_m}{d\ C_L}$ due to the jets = 0.08 is found to be slightly conservative. Measured values on similar models were between 0.04 and 0.05.

I. Interference between Jets

1. Although no experimental data are available at NACA, it is generally agreed that a parallel arrangement of the jet exhaust stacks



is preferable to the convergent arrangement.



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Nacelles and Ducts:

1. Attention of the NACA representatives was drawn to several incorrect assumptions in the nacelle designs of TSESE-41, such as 65-inch total nacelle height for the 65-inch wheel, the excessive width of the NACA nacelle, and the retention of the cooling shroud ahead of the turbine.

2. The NACA has found that there is no need for pressure relief doors in the air intake ducts. On the basis of previous experimental data they estimated an efficiency of 95% for our duct, even at stand-still.

3. The NACA nacelle designs have shown good duct characteristics with one unit inoperative.

4. A report on the optimum lip shape is being released. Comments on the CVAC type nacelle forebody were quite favorable, except that the lip radii and the separator-lip radius should be approximately doubled in order to minimize angle of attack effects, and the effective yawing angle existing during one-unit-inoperative operation.

   NACA representatives agreed warmly with the CVAC air intake duct (Ref. Aero Doc. 109-115) and especially with the conservatism shown in the slow initial expansion close to the intake leading edge where separation due to high angles of attack may occur most readily.

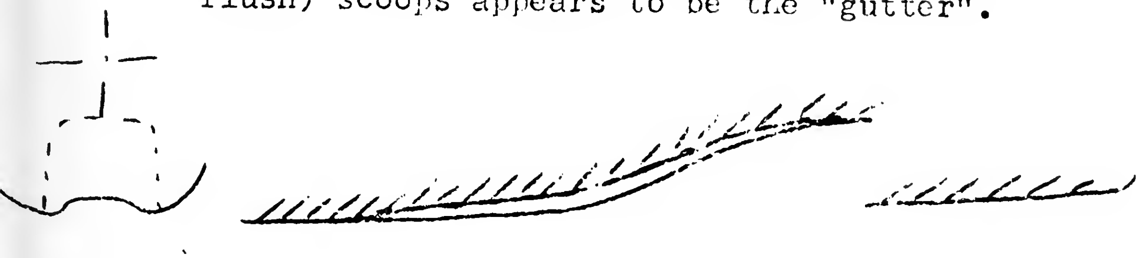
5. NACA recommends that we introduce a rake of hypodermic needle total-head tubes at the location of the blower to determine the ram efficiency of the intake duct, and another set at the jet exit to measure the total drag losses due to the power-nacelle when running HDP's without power.

6. The Cleveland Laboratory is testing various jet exhaust shapes.

IX. Flush Scoops

1. An NACA report on flush scoops is being written; release is expected within approximately 3 months.

An important feature of efficient submerged (or flush) scoops appears to be the "gutter".



X. Effect of Nacelles on Span Load Distribution.

1. NACA has observed a shift in zero lift angle of 1° on the two-dimensional section panels with nacelles, i.e. less than our M.I.T. and Galcit values. Our attempts to reduce this undesirable effect by cambering the nacelles are believed to be steps in the right direction.

2. NACA suggests that we measure pressure on lower flanks of nacelles and fuselage to detect mutual compressibility and interference effects.

I. Lateral Control

No new data available pending the release of generalized NACA wind tunnel data.

I. Dive Recovery Devices

1. No recently released reports exist (see item XI).

2. A P-80a pulled to $C_L = 0.7$ at $M = 0.85$ without using any dive recovery devices.

II. Canopy

1. No canopy-wing interference of compressibility shock fronts is expected in the design range of flying speeds because of the favorable shape of the canopy and its great distance from the wing.

IV. Photographs of Compressibility Shock Fronts.

1. Phenomena observed at CVAC are probably condensation fronts.

2. P-51 flight movies at Wright Field were made by Parsoni. Condensation fronts appear there too. Caltech has a print of the movie.

V. Effects of Wing and Tail Shock Fronts on Control Forces.

1. "Walking" of tail controls on P-51 and P-80 results either from irregular chordwise motion of the shock fronts over the control surface or from the variations in downwash aft of the wing resulting from analogous shock-front movements over the wing (M near 0.80).

2. "Buzzing" of ailerons (approx. 200 to 400 cycles per sec) has also been observed on the P-80a airplane in flight at M = 0.76. (Previous 16' wind tunnel observations had indicated a frequency of 20 cycles per sec.)

I. Air Flow through the Bomb Bay at High Speeds.

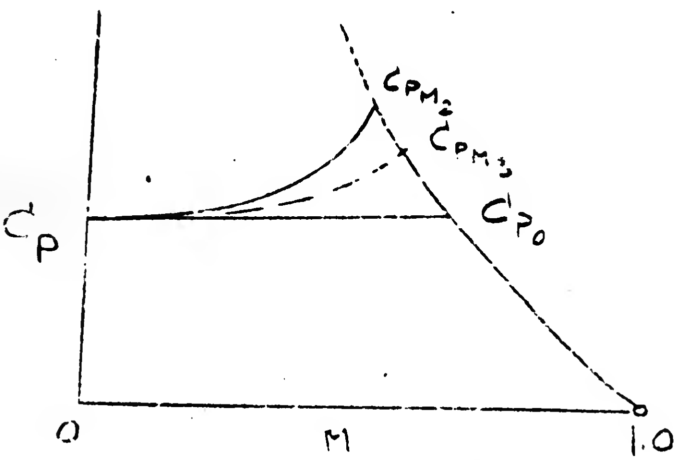
1, NACA suggests that we develop several satisfactory means at low M, before spending high-speed tunnel time on further developments.

2. Mr. Allen has heard from Boeing representatives that some serious troubles have been encountered on the B-29 with bombs tumbling and colliding when released in pairs at high speed. He has seen the newsreels quoted by the Boeing engineers and believes the difficulty to be very real, but does not know what corrective steps Boeing has undertaken.

II. Determination of the Critical Mach Numbers of Three-Dimensional Bodies.

1. In view of the lack of a satisfactory compressibility theory for three-dimensional bodies, Mr. Allen suggests that the critical Mach number of a three-dimensional body be estimated from an increment one-half of the Glauert increment, i.e.

$$C_{P_{M_3}} = \frac{C_{P_0}}{2} + \frac{1}{2\sqrt{1 - M^2}} C_{P_0}$$



I. Availability of NACA Memorandum Reports for AAF and BuAers.

It was remarked that, oftentimes, Memo. Reports for the AAF and BuAer are not issued to CVAC nor are their titles included in the regular NACA lists of reports; that such reports, however, are readily released to CVAC if a specific request, based on information obtained by devious means, is made to the Army or Navy respectively.

NACA representatives are aware of this situation and recommended that CVAC contact Major Jay Auwerter (Army-Wright Field) and Messrs Laudon, Griggs, and Diehl (Navy-BuAer), to have the two agencies make a list of Memo Reports available to CVAC as soon as the reports are released.

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Admitted November 22, 1950.

DEFENDANTS' EXHIBIT D

Consolidated Vultee Aircraft Corporation
General Offices—San Diego, California

December 5, 1944

Effective Control of Stalling Characteristics of Highly Tapered and Swept-Back Wings

By Maurice A. Garbell

Consolidated Vultee Aircraft Corporation

Summary

A tested new method of airfoil selection conceived:

To assist the designer in overcoming present hazardous stalling tendencies on highly tapered and swept-back wings:

To control stall at inception and through progression.

This practicable method eliminates high drag-penalties and other undesirable characteristics which develop with large washout and highly cambered wing tips when employing two controlled sections.

Three controlled sections, one located at the wing root, another at a midspan station, and the third at the wing tip, are connected by straight lines. The principal parameters affecting the maximum section lift coefficient, viz.: the section thickness ratio and camber are chosen to satisfy the section lift coefficients required by the computed span load distribution at the Reynolds numbers of the three spanwise control stations.

Defendants' Exhibit D—(Continued)

The resulting spanwise distribution of maximum lift coefficients permits the designer to exercise close control over the progression of the stall from its inception, and thus reduce washout and camber variation to a minimum. This method also achieves a favorable distribution of critical section Mach numbers along the span. A small but appreciable increase in maximum wing lift coefficients is also obtained.

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Reasons for the Study

The need to overcome hazardous stalling characteristics of highly tapered and swept-back wings has given rise to the present study.

An investigation of the fundamental reasons for these unsatisfactory stalling tendencies reveals that the planform taper of the wing creates two unfavorable effects on the stalling characteristics:

1. The highly tapered planform leads to a deviation from the elliptical span-load distribution in the direction of higher loads at the wing tips for a given wing lift coefficient. Sweep back accentuates this phenomenon. (Fig. 1).

2. The decrease of chord length from the root to the tip reduces the Reynolds number and hence the maximum lift coefficient attainable for a given airfoil.

These two unfavorable developments have been universally counteracted by two measures:

1. Aerodynamic washout, that is, washout of the

Defendants' Exhibit D—(Continued)

zero lift angles, produced by twisting the tip chord with respect to the root chord.

2. The employment of a more highly cambered airfoil at the wing tip than at the wing root.

For manufacturing simplicity the corresponding airfoil stations of the root and tip sections are customarily connected by straight lines. The resulting spanwise variation of aerodynamic washout, camber,

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and thickness ratio is hyperbolic inasmuch as they vary as

$$y = \frac{a + bx}{c + dx}$$

Where: a, b, c, and d are constants depending upon wing geometry x is the spanwise station y is the variable to be determined (aerodynamic washout, camber, and thickness ratio respectively).

Typical spanwise variations are shown in Figure 2.

The principal effect of washout consists of a reduction in the loads at the wing tip and an increase of loads inboard, as shown in Figure 3. The resultant improvement in the stalling characteristics, however, is gained at a penalty in induced drag through the prevalance of positive and negative basic lift over the wing span at zero wing lift (Reference 1).

Washout does not change the section maximum-lift coefficients attainable at the various spanwise stations.

Camber and thickness variations do not affect the spanload distribution (if their slight influence on the

Defendants' Exhibit D—(Continued)

section lift-curve slopes is disregarded), but modify the spanwise distribution of the maximum attainable section lift coefficients.

The straight-line variation of airfoil chord also results in a linear decrease of the Reynolds number from wing root to tip. A nearly linear reduction of section maximum lift coefficients along the span, for a given airfoil section, ensues consequently from the typical maximum lift variation with Reynolds number shown in Figure 4.

A typical spanwise variation in section maximum lift coefficient resulting from the linear fairing of a wing root section and a more highly cambered wing-tip section is portrayed in Figure 5. It is evident

that the line of maximum lift coefficients is concave upward and may even have intermediate stations below the two extremes, because the favorable effect of camber (and thickness) following a hyperbolic law is insufficient to compensate for the unfavorable effect of the linearly disminishing Reynolds number.

As a rule the resulting stall pattern is unsatisfactory for any but the lowest taper ratios and may become critical for taper ratios in excess of 3:1 (see Fig. 5). The stall inception close to the wing tip and the comparatively slow progression of the stall farther inboard produce the most undesirable type of stall, with little or no warning, violent rolling moments, and neutral or unstable pitching moments through the stall.

Defendants' Exhibit D—(Continued)

Any attempt to improve these stalling characteristics by flattening the actual span-load distribution through aerodynamic washout, or raising the curve of the available maximum lift coefficients near the wing tips through adequate amounts of mean-line camber, or by both of these measures, introduces a large drag penalty. In addition, the span-load distribution at the high lift coefficients and Mach numbers occurring during pullouts and steep turns is greatly disturbed by a large spanwise variation of camber. The peak pressure coefficients at high section lift coefficients increase more rapidly over the sections with small camber than over those with large camber, and result in a premature shock stall at the inboard sections, followed by an outboard shift of the air load and a consequent increase in the wing bending moment.

The aforementioned inadequacy of the linearly tapered wing with two controlled sections has led to the development of wings with three controlled sections to permit the designer to obtain the desired

stall inception and progression with a minimum of washout and camber variation.

Definition of Desirable Stalling Characteristics

From the pilot's viewpoint a desirable stall is preceded by a gentle but reliable warning in the form of a mild tail shake some 5-10 mph above stalling speed. The stall should be free from sudden roll, aileron snatch, or severe premature tail buffeting and should be accompanied by a rapid negative

Defendants' Exhibit D—(Continued)

increase of the static longitudinal stability derivative, dCm/dCL.

In order to achieve these desirable characteristics it is advocated that stall separation should start approximately at mid-span, outboard of the horizontal tail (to prevent premature tail shake), and should spread, fairly evenly, inboard and outboard, (Fig. 6). The tail shake then coincides with a ready decrease in the lift-curve slope and the approach to the actual lift-curve peak. The rapid yet gradual spanwise spread of the separated area, simultaneously, prevents the formation of a deep local stall in a chordwise or vertical sense at any section; steep spanwise pressure gradients and hence spanwise cross flow are thereby effectively prevented.

The inboard expansion of the stalled area, aside from producing the desired stall warning, will reduce the downwash at the tail; the increased static longitudinal stability and lowered trim CL provide the nose-down pitching moment which is required for prompt recovery after the stall.

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Stall Characteristics of Wings with Three Controlled Sections ("tri-section wing")

The subject method is based on the use of three controlled sections, at the wing root, another at a mid-span station, and the third at the wing tip, with straight lines connecting the corresponding coordinates.

Defendants' Exhibit D—(Continued)

By judicious selection of the camber and thickness ratios of the three controlled sections it becomes possible to obtain spanwise distributions of maximum section lift coefficients similar to that shown in Figure 7. A comparison of the spanwise distributions of actual and maximum attainable section lift coefficients discloses that the previously postulated requirement of a midspan stall progressing evenly inboard and outboard is met.

A convenient procedure for the selection of the most appropriate parameters (camber and thickness ratio) for the three controlled sections is based on the fundamental information of the variation of maximum lift and zero-lift angle with camber, thickness ratio, and Reynolds number for a given airfoil, required for the respectively selected airfoil family.

A preliminary selection of the three controlled airfoil sections is undertaken, mainly on the basis of past experience. The camber and thickness ratios of several intermediate stations are then determined and the variation of CL max. vs. Reynolds number is plotted for these representative airfoil sections (Fig. 8). Assuming the approximate airspeed at which the stall is expected, the Reynolds numbers of the various spanwise stations are computed and plotted on the CL max. vs. Reynolds Number graph.

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The resulting curve of maximum lift coefficients is then transferred to the graph of CL max. versus span. If the resulting relation between the CL

Defendants' Exhibit D—(Continued)

max. available curve and the spanload distribution is not satisfactory, minor adjustments of the camber, thickness ratios, and the washout will modify the two spanwise distributions until the desired result is obtained.

The variation between maximum lift coefficients and thickness ratio shows a certain peculiarity which can be employed to good advantage. Most airfoil families reach their absolutely highest CL max. at a thickness ratio between 12 and 16 per cent. Thickness ratios greater or lesser than the optimum value result in lower maximum lift coefficients. Consequently, if a thickness lesser than optimum is used for the wing tip, where the load is greatly reduced from its peak value, the optimum airfoil thickness can be located at the spanwise station a small distance inboard of the wing tip where the highest load is reached (Fig. 9).

Wind Tunnel Testing for Stalling Characteristics

Wind-tunnel testing on small-scale models for the prediction of the full-scale stalling characteristics is generally not entirely satisfactory because it is extremely difficult to reproduce the full-scale Reynolds number without exceeding the full-scale Mach number. This is particularly disconcerting when testing in small, atmospheric tunnels during the preliminary-design stage of a new-type aircraft, at which phase accurate data for the estimation of the full-scale stalling characteristics are most urgently required.

Defendants' Exhibit D—(Continued)

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Some assistance, at least, on this perplexing problem can be gained from the CL max. vs. Reynolds Number graph, where model Reynolds numbers are used instead of full-scale values.

No general rule on the comparative character of the stalling characteristics at model and full-scale can be advanced but it is recommended that a prediction of the model stalling characteristics be made prior to the wind-tunnel test not only to test the accuracy of the method, but also to uncover the existence of any unforseen interference factors on the stall characteristics.

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Conclusion

The adoption of a third controlled airfoil section near mid-span permits the attainment of any desired stall characteristics by eliminating the localized deep stalls over the outboard panels.

A desirable apportionment of spanwise lift distribution at relatively high lifts and Mach numbers can be determined for given stall characteristics, because a satisfactory stall can be obtained with a smaller spanwise variation of camber.

The method has been successfully tested on wings with taper ratios up to 4:1 and leading edge sweepbacks up to 15°. Because of military restrictions the visual demonstration of stall characteristics on a wind-tunnel model must be limited to photographs of a non-confidential research wing with taper ratio 3:1 which is, however, fully representative of wings

Defendants' Exhibit D—(Continued)

with higher taper ratios and greater sweepback. The airfoils used are NACA 2518, 3515 and 4512, respectively (Ref. 2). No aerodynamic washout is incorporated. A theoretical comparison of the stalling characteristics of this wing and a wing with straight line fairing between a 2518 root airfoil and a 4512 tip airfoil (no aerodynamic washout) is shown in Figure 10. It is of significance that the stall of the "tri-section wing" begins at a wing lift coefficient of 1.5 against a stalling lift coefficient of 1.4 in a conventional straight-line faired two-section wing. Photographs 1 to 5 substantiate the concurrence of estimated and experimentally obtained characteristics of the "tri-section wing."

Page 10 of 14

References

1. Determination of the Characteristics of Tapered Wings by Raymond F. Anderson NACA Technical Report No. 572, 1933.

2. The Characteristics of 78 Related Airfoil Sections from Tests in the Variable—Density Wind-Tunnel by Eastman N. Jacobs,
Kenneth E. Ward, and
Robert W. Pinkerton
NACA Technical Report No. 460, November, 1933.

DEC 20

FIG. 1

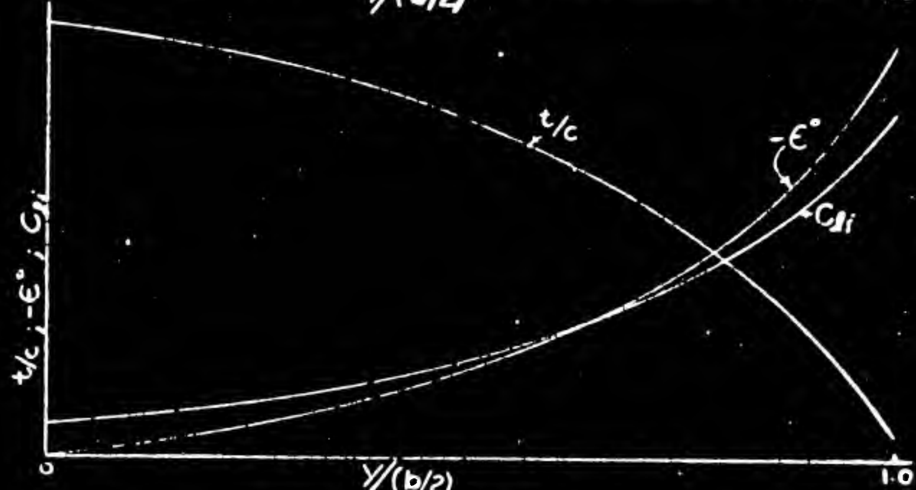


FIG. 2

FIG. 3

BY
CHECKED
APPROVED

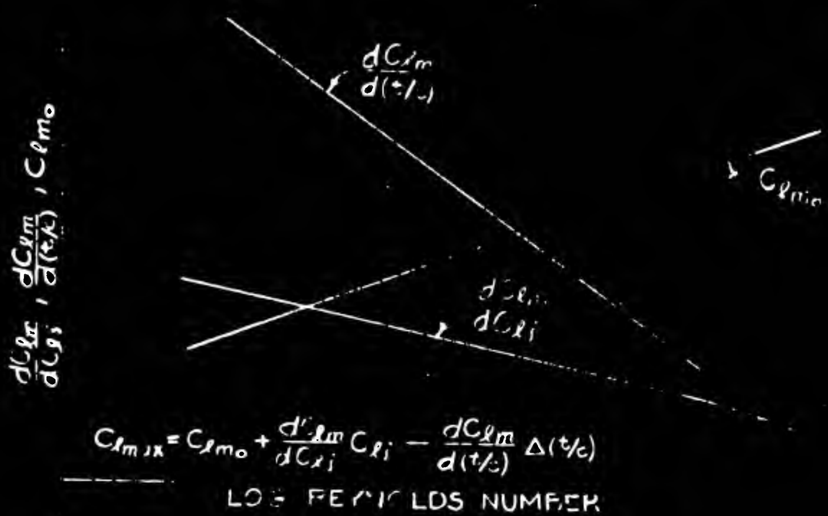


FIG. 4

FIG. 5

FIG. 6

MODEL AIRPLANE REPORT NO

FIG. 7

FIG. 8

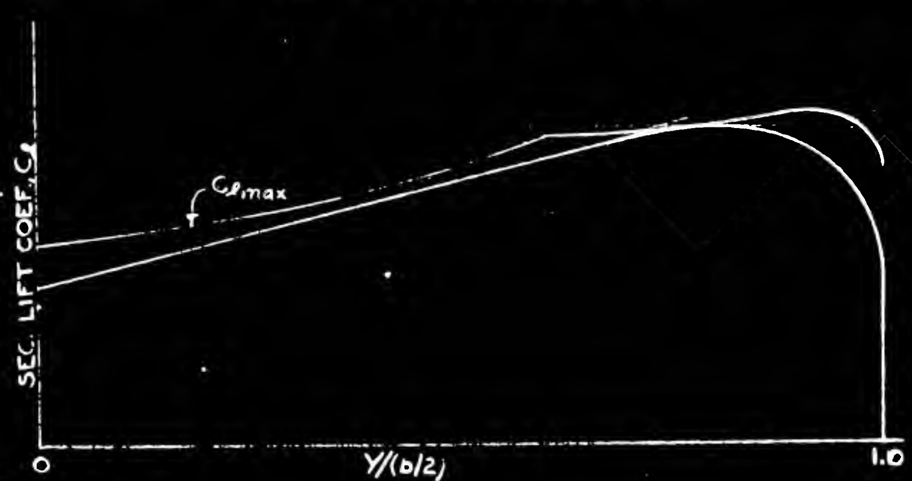


FIG. 9

BY

CHECKED

APPROVED

MODEL AIRPLANE REPORT NO

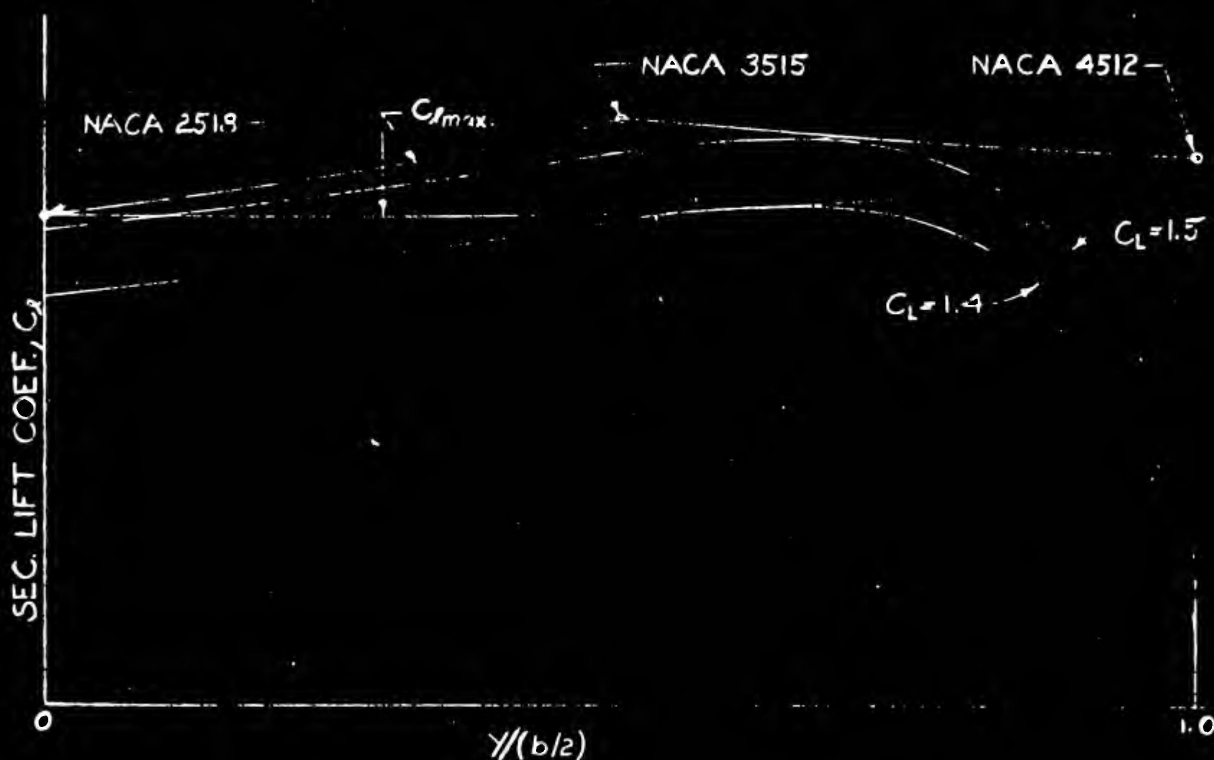


FIG. 1C

[illegible]tted [illegible], 19[illegible]0.

BY

CHECKED

APPROVED

DEFENDANTS' EXHIBIT E

Consolidated Vultee Aircraft Corporation
General Offices, San Diego, California

19 December, 1944

Mr. D. A. Hall
Mr. M. A. Garbell

Disclosure of Method of Effective Control of Stalling Characteristics of Highly Tapered and Swept-Back Wings.

Enclosed is a copy of my paper on "Effective Control of Stalling Characteristics of Highly Tapered and Swept-Back Wings" for your information and file.

Please consider this paper an official disclosure of invention. I shall be glad to complete the disclosure with any additional material that may be requested by you.

M. A. GARBELL.

[Stamped]: Patent Dept., Dec. 20, 1944. Consolidated Vultee Aircraft Corporation.
MAG:lm

Admitted November 22, 1950.

DEFENDANTS' EXHIBIT F

Consolidated Vultee Aircraft Corporation
San Diego Division, San Diego, California

January 8, 1945

Dr. Maurice A. Garbell
R. Evers

Your Disclosure on Stall Characteristics on Variable Section Wings

(a) D. A. Hall verbal request Jan. 1, 1945

It has been brought to our attention that some additional information would be desirable to further clarify your subject disclosure.

Mr. D. A. Hall has requested that, if available, the following data be sent to him:

(a) Curve showing reduction of drag coefficient (Cd) by your method over the conventional design.

(b) A tabulation of symbols used in the disclosure.

(c) Copies of N.A.C.A. references.

Mr. Hall should also be advised if you have received the information you requested from Vultee Field.

R. EVERS.

RE:mh
cc: R. Evers
Dev. Engr. File

Admitted November 24, 1950.

DEFENDANTS' EXHIBIT G

District Court of the United States, Southern District of California, Central Division

Civil Action No. 10930-Y

MAURICE A. GARBELL, INC., a California Corporation, and GARBELL RESEARCH FOUNDATION, a California Corporation,

Plaintiffs,

vs.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION, a Delaware Corporation, and AMERICAN AIR LINES, INC., a Delaware Corporation,

Defendants.

STIPULATION #11

It is hereby stipulated subject to proof of error that the appended "Exhibit 125" is a reproduction of pages 8 and 9 of Volume 8 of a printed publication "L'Aquilone" containing an article entitled "Tre nuovi veleggiatori italiani per il 1938" published and issued by Editorial Aeronautica in Rome, Italy, in the year 1938, and that "Exhibit 125a" is a translation of said article (subject to correction if any error is contained therein), and that said "Exhibit 125" may be used in evidence with the same force and effect as an original, subject to any

Defendants' Exhibit G—(Continued)

objection which may be made thereto as irrelevant or immaterial when offered in evidence, viz.;

LYON & LYON,
/s/ FREDERICK W. LYON,
Attorneys for Plaintiffs.

/s/ ROBERT B. WATTS,
/s/ FRED GERLACH,
Attorneys for Defendants.

Exhibit 125a

From L'Aquilone, Jan. 16, 1938, pp. 8 & 9

Translation from: Italian WB:GS

Three New Italian Soaring Gliders for 1938

The great advance which has recently been experienced by Italian gliding in the 15th Year of the Fascist Regime, as a result of the interest shown by the executives of the R.U.N.A., has placed the problem of soaring gliders on the agenda. In Asiago, we saw Italian planes which had been constructed or else designed at least 3 to 4 years ago, as well as the German soaring gliders brought from Cattaneo, namely the "Condor I" and the "Hutter" 17, which no longer represent the last word in the construction of gliders.

This situation was well understood by the Gliding Research and Experimental Center of the Royal Polytechnicum and the G.U.F. of Milan and also by the Societa Aeronautica Lombarda which, as is well-known, has up to now supplied almost all the

Defendants' Exhibit G—(Continued)

gliders to the motor-less flight schools of the Party. Between the two organizations, the one of a scientific technical nature and the other of a manufacturing nature, a fruitful agreement has been entered into in accordance with which the S.A.L. will greatly assist the Center in the construction of its two models, while the latter agreed to grant licenses for mass production.

In accordance with this agreement, which shows a characteristically Fascistic spirit of cooperation, and for the purpose of finally giving Italy the models which the ability of the Italian pilots merits, there were rapidly brought out the three models which were to represent the three classical categories of the high-gliding school, namely the performance-type glider, the secondary-type glider and the primary-type glider. There should be mentioned the extremely short time of construction: The "Pinguino GP. 1," a high-class soaring glider, was designed and constructed in 150 work days (during the Asiago rally, the work was interrupted due to the absence of the designers); the "Alcione BS 28" and the "Asiago GP 2" were both born in 100 days, counting from the first rough sketch to the flight test.

"L'Aquilone" has already related ("The Birth of the Pinguino" and "At the Salon") the story of the construction of the Pinguino and of the "Asiago" and there will now be described briefly (as we ourselves have seen it) the testing of the 3 planes. We now have a clear idea as to how these

Defendants' Exhibit G—(Continued)

3 planes are made, how they were born and how they will be used.

Let us follow the chronological order of the creation of the 3 planes; first of all, the "Pinguino GP 1," which did not see the light of day in the subterranean darkness of the Milan Polytechnicum.

The "Pinguino G. P. 1"

The external lines of the "Pinguino" are those characteristic of M central wing soaring gliders ("Rhonsperber," "Tulak," etc.). The main technical specifications are:

Wing span	13.30	meters
Length	6.50	"
Wing surface	15.20	m2
Aspect ratio	15	
Deadweight	170	kgs.
Useful load	80	"
Total weight	250	"
Wing loading	15.2	kg/m2
Coefficient of strength	9	
Minimum velocity of descent in m/sec.	0.69	
Gliding angle	1:25.3	

The wing is completely of the cantilever type. The plan of the wing is rectilinear in the central portion and tapers towards the tip. In the central portion, the dihedral of which is 6°, there has been used the Gottinga G 535 profile which is constant up to the bend. At the tip, however, the N.A.C.A. 23012 profile is used. The course of the profile in

Defendants' Exhibit G—(Continued)

the tapered part of the wing is gradual and linear. The geometrical warping (i.e., of the reference chords) of the two airfoils is 0, but in view of the difference between the conventional reference chords in the Gottinga and N.A.C.A. systems, the aerodynamic warping attains a value of about 3°. In the first three ribs at the root of the wing, the G. 535 profile is not, however, maintained constant, but passes with a parabolic course into an ideal N.A.C.A. 0015 profile which, as is well known, is symmetrical. The connection between wing and fuselage is effected almost automatically, which greatly improves the lift distribution on the wing in the vicinity of the fuselage.

The wing is of the monospar type, with a small false rear spar. The main spar is of the box type, consisting of upper and lower cap-strips connected with each other by means of the two plywood side walls. The cap-strips are of spruce plywood, that is to say they consist of many strips of a height of about 1 cm. which are glued together. In this manner, the spar is not only much stronger than a spar made of a single piece, but it is also possible to construct the spar without connection to the bend of the M since the use of glued plywood does away with the internal stresses coming from the bending of the individual strips.

The leading edge contributes greatly to the resistance, withstanding practically all the torsional forces. It is therefore covered with birch plywood of a thickness of 10/10, 15/10 and 20/10 mm. Need-

Defendants' Exhibit G—(Continued)

less to say, all the plywood used is first-quality wood, approved by the G.A. In order to maintain the form of the torque tube which is the leading edge, false ribs are interspersed between the ribs, the 30 cm. distance between which appeared excessive for this purpose. All of the ribs are of domestic poplar of first-class quality, having normal panel-work and reinforced with plywood gussets of a thickness of 10/10.

The aileron is of a single piece controlled by two levers, but at the present time it has been divided into two parts of differential action in order further to increase the efficiency. The transverse control has a differential of about 1:2.5 and therefore one aileron rises about 2.5 times more than the other is lowered. As a matter of fact, it is known that in order to obtain equal values of increase or decrease of lift, the aileron must have a greater travel upward than downward. Furthermore, an excessive lowering of the aileron is harmful in that the lowering, in addition to increasing the lift of the wing, also increases the resistance to forward motion. If, for example, we give "contrary ailerons" while banking, for the purpose of straightening the ailerons, there takes place a braking of the inner wing and therefore an action which tends to maintain the plane in the bank. This entire reasoning has brought about the idea of applying differential control to the ailerons.

The transmission of the torsional forces from the leading edge to the fuselage occurs along the diag-

Defendants' Exhibit G—(Continued)

onal which transmits them to the rear connection of the wing to the fuselage. The metal connections are of carbon and chrome molybdenum steel and the pins are of chrome molybdenum steel.

It is already known that on each pin there acts about 19,200 kgs. compression or tension, which justifies the use of steel of the highest strength.

On the upper surface of the wing, there is located a CVV type flap of 600 sq. cm. surface. The purpose of this flap is to increase the velocity of descent from 0.70 to about 2 meters per second and to change the gliding angle to about 1:10. The flap consists of a duraluminum plate set normally to the upper surface of the wing. By means of a simple mechanism consisting of a few curved levers, a cable and 7 rollers, the movement of the flaps is controlled by a lever located beneath the instrument board. It suffices for the pilot to pull this lever in order to elevate the two flaps. Two torsion springs return the flaps into the rest position as soon as the pulling on the lever ceases. The CVV flaps have proven extremely efficient right from the first flight. The progress realized, as compared with the old Jacobs flaps which adhered to the wing along an edge, is remarkable. The disturbing effect is considerably greater but at the same time more regular. In no case was there noticed any vibration or shaking of the tail, which is so troublesome in other gliders. The efficiency of the CVV flap is of course not as great as that of the double split flaps of the Jacobs type, which however cost much more on

Defendants' Exhibit G—(Continued)

account of the greater mechanical complication. In any event, the results obtained up to the present time are very encouraging.

The fuselage is of ovoid section generated by circular arcs. While in the rear part of the fuselage, there are three circular arcs, leaving one sharp edge below; in the rear part, the shape consists of four connected arcs. With a somewhat simpler design, there is thus obtained an excellent section. The sharp angle keel which is present in the rear part of the fuselage has an important stabilizing action, especially during sustained flight. As a matter of fact, it retards and hampers the side slip.

The fuselage consists of six spars and twenty frames. However, the main purpose of these members is to maintain the shape of the fuselage intact inasmuch as the resisting member is constituted by the plywood covering. We thus have a monocoque structure.

The elevator consists of a fixed plane entirely of the cantilever type connected to the fuselage by means of four bolts and a movable unbalanced plane. The control of the latter is effected by means of a lever on the inside. Not even the rudder is aerodynamically compensated.

The cockpit is very commodious. The adjustable seat perfectly fits the shape of the human body. The pedals consist of two wooden pedals hinged at the bottom. The cowling is completely transparent and offers optimum visibility in all directions, even rearward.

Defendants' Exhibit G—(Continued)

A normal ash skid covered with a thin strip of sheet steel and made resilient by rubber absorbers, absords the landing shocks. The tail skid consists of a strip of duraluminum sheet metal below the rear nose.

In the next issue, we shall publish the description of one of the other two soaring gliders.

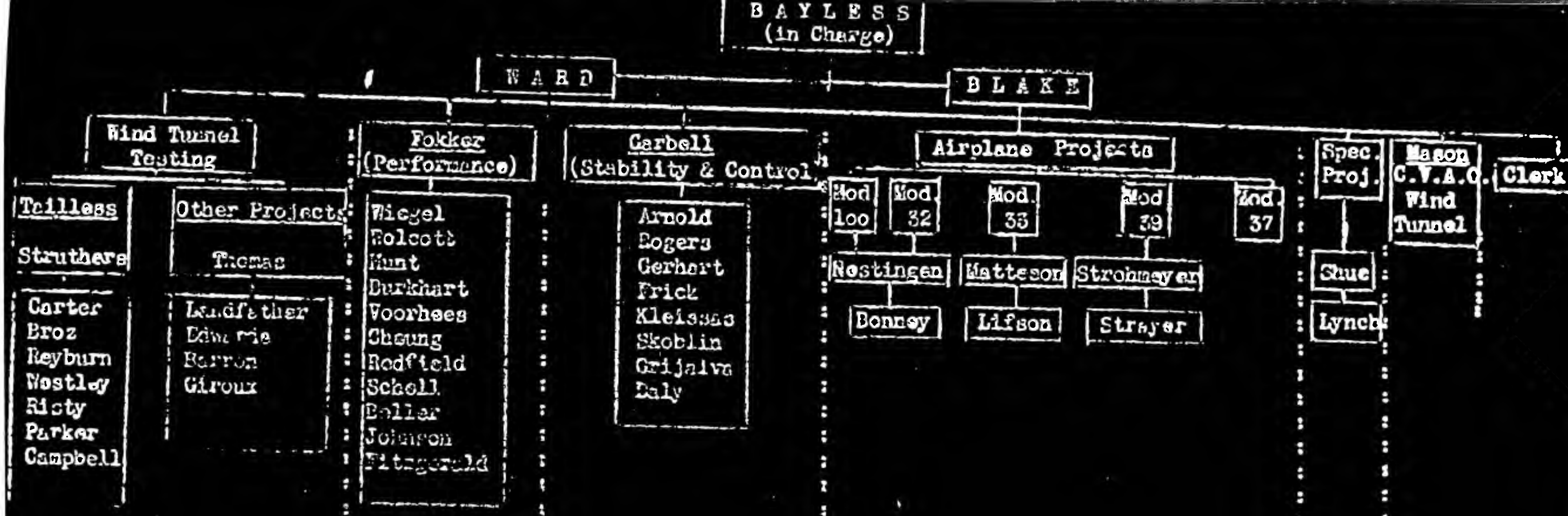
Translation of Captions

(A) These three photographs show the "Pinguino" just after assembly, top view; in the center there is shown the statical testing of the wing, and at the bottom there is shown the glider in flight.

(B) A view of the frame of the fuselage of the "Pinguino" during construction.

(C) The frame of the right half-wing of the "Pinguino."

Admitted November 22, 1950.



April 6, 1944.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION
LINDBERGH FIELD - SAN DIEGO, CALIFORNIA.

Defts Ex 117

944

DEFENDANTS' EXHIBIT I/

LINDBERGH FIELD SAN DIEGO, CALIFORNIA /C+

April 4, 1944.

RESPONSIBILITY

[illegible] important decisions with respect to aerodynamic design, selection [illegible] engines, propellers, etc; all performance, wind tunnel test programs, [illegible]d tunnel models; and all changes to airplanes under flight test shall [illegible] approved in writing by the Engineer in Charge except that Mr. Ward, [illegible]ing as an Assistant to the Engineer in Charge, shall approve the work [illegible]e by the Airplane Projects and Special Projects personnel and Mr. [illegible]ke, acting as an Assistant to the Engineer in Charge, shall approve [illegible] work done by the Wind Tunnel personnel.

[illegible]d Tunnel Testing -
- Wind tunnel models.
- Wind tunnel tests.

[illegible]formance -
- All performance
- FOR NEW DESIGNS
- (a) Wing Area
- (b) Aspect Ratio
- (c) Tail Area
- (d) Fuse. Shape
- (e) Nac. Shape
- (f) Turret Inst.
- (g) Protuberance Des.
- (h) Surface Cond.
- (i) Leakage
- (j) Misc. Char. Affecting Drag
- (k) Engine Selection and Establishment of BHP, SFC, etc.
- (l) Prop. Selection and Eng. Gear Ratio
- (m) Turbo Selection
- (n) Flap Selection
- (o) Landing Gear Tread Wheelbase and Ground Angle
- (p) Satisfactory Attitude of Fuselage for Various Flight Conditions
- For EXISTING DES.
- All Characteristics Affecting Drag and Performance
- Satis. Eng. Cooling New and Existing Designs

Clerk -
- (a) All scheduled records and status reports
- (b) Time record
- (c) Files

Stability and Control -
- (1) Satisfactory stability, control and flying characteristics in flight and in wind tunnel tests
- (2) Wind tunnel reports, test programs and test predictions
- (3) Aero. data for structural analysis
- (4) FOR NEW DESIGNS
  - (a) Airfoils
  - (b) Wing, tail and control surface, geometry and design
  - (c) Wing and tail incid.
  - (d) Tail Length
  - (e) Flap design
  - (f) C.G. limits for adequate stability and control

Airplane Projects -
- (1) All specific flight predictions
- (2) Flight test programs
- (3) Flight prediction and characteristics summary report (60 days prior initial flight of new design)
- (4) Engineering and Shop Liaison to assure that all aerodynamic characteristics are maintained as planned.

Special Projects -
- (1) Prediction of compressibility effects

CVAC Wind Tunnel -
- (1) Expedite and coordinate CVAC Wind Tunnel Project

[signature] III

[signature]

161

DEFENDANTS' EXHIBIT J

CONSOLIDATED AIRCRAFT CORPORATION — San Diego, California

Employment Division

Industrial Relations Department

ASSIGNMENT TO EMPLOYMENT

ame.. Sex .... Clock No. 6-1801

Shift

1 ....

partment.. Rate 2 ....

3 ....

assification.. Code 67531

arting Date.. Hour ....

marks:

Hired by.. Date..........

m 609

EXH 111

CONSOLIDATED VULTEE AIRCRAFT CORPORATION SAN DIEGO, CALIF.

CHANGE IN EMPLOYEE'S STATUS

NAME Garbell, M. A. EFFECTIVE DATE 8-21-43 TIME A M / P M

	FROM	TO
DEPARTMENT	Dev. Eng.	
CLOCK NUMBER	1-61-1082	
RATE & SHIFT	(175. SM) 162.40 BW	162.40 BW

JOB TITLE GRADE AND CODE NUMBER — FROM Aero. Engineer A 7531 — TO same

REASON: Order 435

RATE CHANGE ☐ JOB TITLE CHANGE ☐ SHIFT CHANGE ☐ INTER-DEPT. TRANSFER ☐ RECORD CORRECTION ☐

SIGNED C A Phillips DEPARTMENT FROM SIGNED DEPARTMENT TO

WAGE REVIEW

APPROVED

APPROVED

MANAGEMENT

APPROVED Ree

APPROVED R M Andrews

ACCOUNTING

ADD M.M.

TAB TAB AUG 21 '43

PAYROLL

FORM 1039A

ACCOUNTING DEPT. COPY

EXH 112

CONSOLIDATED VULTEE AIRCRAFT CORPORATION — SAN DIEGO DIVISION

CHANGE IN EMPLOYEE'S STATUS

FSP

NAME: M. A. Garbell — EFFECTIVE DATE 5-1-44 — TIME A.M. P.M.

	FROM	TO
DEPARTMENT	Dev. Engr.	Same
CLOCK NUMBER	661-1-379745 F	Same
RATE & SHIFT	247.50 SM FSP 1st	267.50 SM FSP 1st

JOB TITLE GRADE AND CODE NUMBER: FROM Group Engineer 1160; TO Same

REASON: Six Months Review

MALE	FEMALE	FULL TIME	PART TIME	ADULT	MINOR

RATE CHANGE [X] — JOB TITLE CHANGE [ ] — SHIFT CHANGE [ ] — INTER-DEPT. TRANSFER [ ] — RECORD CORRECTION [ ]

SIGNED M Hall — DEPARTMENT FROM

SIGNED R. M. Andrews — DEPARTMENT TO

WAGE REVIEW — MANAGEMENT — ACCOUNTING

APPROVED — APPROVED H. A. Sutton — ADD

APPROVED — APPROVED C. G. Long — TAB — PAYROLL

FORM

ACCOUNTING DEPT. COPY

EXH 118

CONSOLIDATED VULTEE AIRCRAFT CORPORATION — SAN DIEGO DIVISION

1868 CHANGE IN EMPLOYEE'S STATUS

FSP

NAME: Garbell, M. A. — EMPLOYEE NUMBER 379745 — PAYROLL FSP

DATE ISSUED 12-21-44 — TIME A.M. P.M. — DATE EFFECTIVE 2-1-45 — TIME A.M. P.M.

Retro 1-1-45

	FROM	TO
DEPARTMENT	Dev. Engr. 661-1-	Same
RATE & SHIFT	(267.50)267.50SM FSP1st	(285.)285. SM FSP 1st

JOB TITLE GRADE AND CODE NUMBER: FROM Group Engineer 1160; TO Design Specialist A 2633

REASON: Adj. toward minimum

P&A Conv.

MALE	FEMALE	FULL TIME	PART TIME	ADULT	MINOR
X		X		X	

RATE CHANGE [X] — JOB TITLE CHANGE [ ] — [ ] — INTER-DEPT. TRANSFER [ ] — RECORD CORRECTION [ ]

SIGNED M Hall — DEPARTMENT FROM

SIGNED — DEPARTMENT TO

WAGE REVIEW — MANAGEMENT — ACCOUNTING

APPROVED P.A. — APPROVED B. W. Sheahan — ADD

APPROVED — APPROVED C. T. Reid — TAB — PAYROLL

FORM 1038C

ACCOUNTING DEPT. COPY

F S P COP

CONSOLIDATED VULTEE AIRCRAFT CORPORATION SAN DIEGO, CALIF

CHANGE IN EMPLOYEE'S STATUS

NAME Garbell, Maurice A EFFECTIVE DATE 11-1-43 TIME A M P M

	FROM	TO
DEPARTMENT	Dev. Engr. 1-61	Same
CLOCK NUMBER	1-61-1082	1-61-155
RATE & SHIFT	162.40 BW	247.50 SM FSP

JOB TITLE GRADE AND CODE NUMBER FROM Aero. Engineer "A" 7531
TO Design Engineer 7610

REASON: Merit Increase NO RED BADGE

RATE CHANGE [X] JOB TITLE CHANGE [X] SHIFT CHANGE [ ] INTER-DEPT. TRANSFER [ ] RECORD CORRECTION [ ]

SIGNED C. A. Phillips SIGNED
DEPARTMENT FROM DEPARTMENT TO

WAGE REVIEW
APPROVED
APPROVED

MANAGEMENT
APPROVED
APPROVED R. M. Andrews

ACCOUNTING OCT 30 '43 ADD
ADD
TAB TAB. NOV 02 43
PAYROLL

FORM 1039A

ACCOUNTING DEPT. COPY

EX 114

CONSOLIDATED VULTEE AIRCRAFT CORPORATION SAN DIEGO DIVISION

CHANGE IN EMPLOYEE'S STATUS

FSP

NAME Garbell, M.A. EFFECTIVE DATE 3-16-44

	FROM	TO
DEPARTMENT	Dev. Engr.	Same
CLOCK NUMBER	661-379745 F	Same
RATE & SHIFT	FSP 1st	Same

JOB TITLE GRADE AND CODE NUMBER FROM Aerody. Engr. A 2533
TO Group Engineer 1160

REASON: S.S.U. Conv.

~~MALE~~ ~~FEMALE~~ FULL TIME PART TIME ~~ADULT~~ ~~MINOR~~

RATE CHANGE [ ] JOB TITLE CHANGE [X] SHIFT CHANGE [ ] INTER-DEPT. TRANSFER [ ] RECORD CORRECTION [ ]

SIGNED CA Phillips SIGNED
DEPARTMENT FROM DEPARTMENT TO

WAGE REVIEW
APPROVED
APPROVED

MANAGEMENT
APPROVED
APPROVED R. M. Andrews

ACCOUNTING MAR 14 '44
ADD
TAB
PAYROLL

FORM 1039A

ACCOUNTING DEPT. COPY

EXH 115

Admitted November 22, 1950.

DEFENDANTS' EXHIBIT K

Contract No. W 535 ac-24664
(6731)

Contract
(Supplies)

MW:RH

ANMB Preference AA-1 & A-1-A
Allocation Classification System Symbols: USA-1.00

War Department
(Department)

Vultee Aircraft, Inc.
(Contractor)

Contract for 400 A-35B Airplanes, Static Test Airplane, Spare Parts and Data. Amount, $34,034,840.00.

Place: Army Air Forces, Materiel Center, Wright Field, Dayton, Ohio.

The Finance Officer, U. S. Army, Wright Field, Dayton, Ohio, is designated as the officer to make payments in accordance with this contract. The supplies and services to be obtained by this instrument are authorized by, are for the purpose set forth in, and are chargeable to the Procurement Authorities listed hereon, the available balances of which are sufficient to cover the cost of the same.

AC 2312
P 12-09
A 0705-23 . . . .$24,339,000.00 Class. 01-A
AC 2382
P 82-09
A 0705-23 . . . . 9,695,840.00 01-Q
$34,034,840.00 AAF Stock No. 0103

Defendants' Exhibit K—(Continued)

AFP: 194465

This contract supersedes Letter Contract Special Form dated January 16, 1942.

Approval recommended: September 22, 1942.

/s/ O. P. ECHOLS,
Major Gen., U.S.A., Commanding General, Materiel Command.

Approved: Sep. 22, 1942.

By direction of the Secretary of War, under the provisions of the First War Powers Act, 1941, and Executive Order No. 9001, December 27, 1941.

/s/ PHILLIPS W. SMITH,
Col., A.U.S., Special Representative of the Under Secretary of War.

Article 51

Approval.—This contract shall be subject to the written approval of the Secretary of War or such individual as said Secretary may designate and shall not be binding until so approved. The date of such approval shall be deemed to be the true date for the purpose of determining all times of performance.

Article 52

Alterations.—The following changes were made in this contract before it was signed by the parties hereto: Articles 15, 15A, 16, 16A, 17 to 52, inclusive, on pages 4a, 4a-1, 4a-2, 4a-3, 4a-4, 4a-5, 4b to 4n, 4n-1, 4n-2, 4n-3, 4n-4, 4n-5, 4o to 4s, inclusive, added, all as approved by the Director of the Bureau of

Defendants' Exhibit K—(Continued)

the Budget and/or the Under Secretary of War. The letter "(a)" inserted after the heading "Taxes" in the first line of Article 29 and paragraph (b) added thereto.

In Witness Whereof, the parties hereto have executed this contract as of the day and year first above written.

THE UNITED STATES OF AMERICA

By /s/ JOSEPH E. DERHAM,
Lt. Colonel, Air Corps, Contracting Officer, U. S. Army.

VULTEE AIRCRAFT, INC.,
Contractor,

By /s/ V. C. SCHORLEMMER,
Vice-Pres.,
Downey, California.
(Business address)

Two witnesses:

/s/ GLORIA WEAVER,
/s/ BETTY BROTHER.

I, T. C. Sullivan, certify that I am the Secretary of the corporation named as contractor herein; that V. C. Schorlemmer, who signed this contract on behalf of the contractor, was then Vice President of said corporation; that said contract was duly signed for and in behalf of said corporation by

Defendants' Exhibit K—(Continued)

authority of its governing body, and is within the scope of its corporate powers.

[Corporate Seal]

/s/ T. C. SULLIVAN.

I hereby certify that, to the best of my knowledge and belief, based upon observation and inquiry, ..., who signed this contract for the ...., had authority to execute the same, and is the individual who signs similar contracts on behalf of this corporation with the public generally.

........................,
Contracting Officer.

W 535 ac-24664

Defendants' Exhibit K—(Continued)

Supplemental Agreement No. 1
to
Contract W 535 ac-24664

Contractor: Vultee Aircraft, Inc.
Vultee Field, California

X

X

X

X

X

Approval Recommended: December 15, 1942.

/s/ O. P. ECHOLS,
Major Gen., U.S.A., Commanding General Materiel Command.

Approved: Dec. 17, 1942.

By direction of the Secretary of War, under the provisions of the First War Powers Act, 1941, and Executive Order No. 9001, dated December 27, 1942.

/s/ PHILLIPS W. SMITH,
Lt. Col., Ord. Dept.

ALBERT J. BROWNING,
Colonel, General Staff Corps, Special Representative of the Under Secretary of War.

Defendants' Exhibit K—(Continued)

In Witness Whereof, the parties hereto have executed this Supplemental Agreement No. 1 as of the day and year first above written.

THE UNITED STATES OF AMERICA,

By /s/ JAMES W. SHOCKNESSY,
Capt., A.C.,
Contracting Officer,

WM. MITCHELL,
Captain, Air Corps, U. S. Army, Contracting Officer.
(Official Title)

VULTEE AIRCRAFT, INC.,
(Contractor)

By /s/ DAVID G. FLEET,
Executive Vice-President,
Vultee Field, California.
(Business Address)

Two Witnesses:

/s/ E. LAESAKU,

/s/ C. W. CROCKER.

I, O. R. Stocke, certify that I am the Assistant Secretary of the corporation named as Contractor herein; that David G. Fleet, who signed this Supplemental Agreement on behalf of the Contractor, was then Executive Vice-President of said corporation; that said Supplemental Agreement was duly signed for and in behalf of said corporation by

Defendants' Exhibit K—(Continued)

authority of its governing body, and is within the scope of its corporate powers.

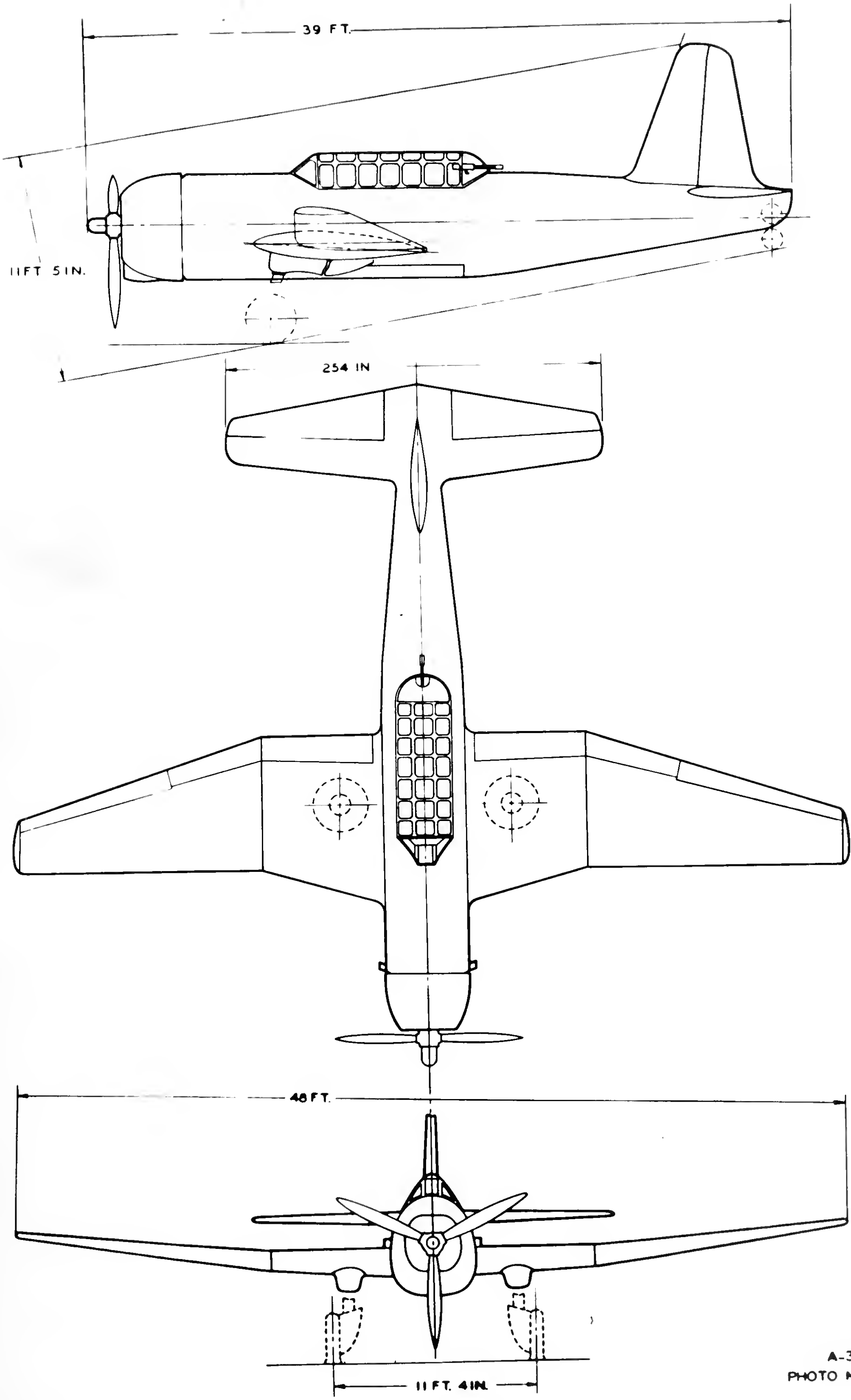
[Corporate Seal]

C. R. STOCKE.

I hereby certify that, to the best of my knowledge and belief, based upon observation and inquiry, ..., who signed this Supplemental Agreement for Vultee Aircraft, Inc., had authority to execute the same, and is the individual who signs similar contracts on behalf of this corporation with the public generally.

.........................,
(Contracting Officer)

Page 8 of Supplemental Agreement No. 1 to Contract No. W 535 ac-24664.



A-35B-VN
PHOTO NO 14581

8 1 4

5

FOR OFFICIAL USE ONLY

AIR PUBLICATION 2024A
VOLUME I

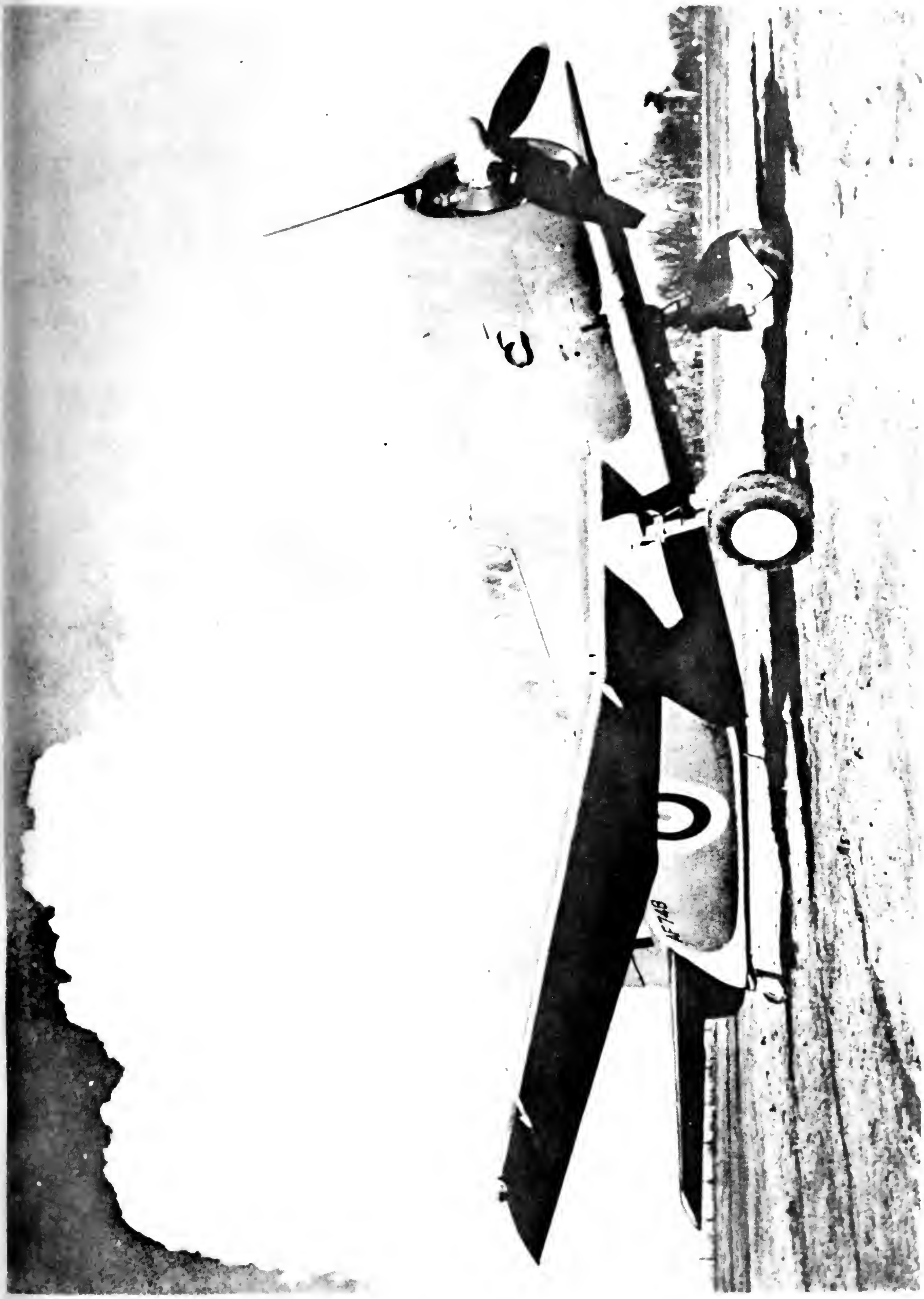
VENGEANCE I AEROPLANE

WRIGHT GR-2600-A5B-5 ENGINE

AIR MINISTRY

B.A.C./3.42/VULTEE

JUNE 1942



FRONTISPIECE

Defendants' Exhibit K—(Continued)

Air Publication 2024A
Vol. I

Leading Particulars

Type	Two-seater, single engined, low wing, land monoplane
Duty	Day and night dive bombing

Principal Dimensions

(Airplane in flying attitude unless otherwise stated)

Span	48 ft. 0 in.
Length (Overall)	40 ft. 0 in.
Height (Over radio mast)	12 ft. 0.69 in.
Length (Tail wheel on ground)	39 ft. 4.3 in.
Height (Over propeller tip, tail wheel on ground)	14 ft. 6 in.

Wing

Airfoil Section:	
At Wing Root	NACA 14516-64
At Outer Panel Joint	NACA 14516-64
At Tip	NACA 20509-64
Chord at Fuselage Centerline	10 ft. 6 in.
Chord at Outer Panel Joint	7 ft. 6 in.
Chord at Tip	3 ft. 6 in.
Incidence	0°
Dihedral measured on chord plane of Inner Panel	1° 33′36″
Dihedral measured on chord plane of Outer Panel	7°
Sweepback at leading edge of Inner Panel	16° 10′52″
Sweepback at leading edge of Outer Panel	0°

818

11

Vultee

MODEL 72

REPORT 1793 Part I

DATE 9-16-41

TITLE

WING REPORT

SUBMITTED UNDER

Contract No. 557

PREPARED BY: CASWELL

BANKERD

GROUP: Stress

REFERENCE:

CHECKED BY: ZALLER

COLE

APPROVED BY: Clements

NO. OF PAGES

NO. OF DIAGRAMS

REVISIONS

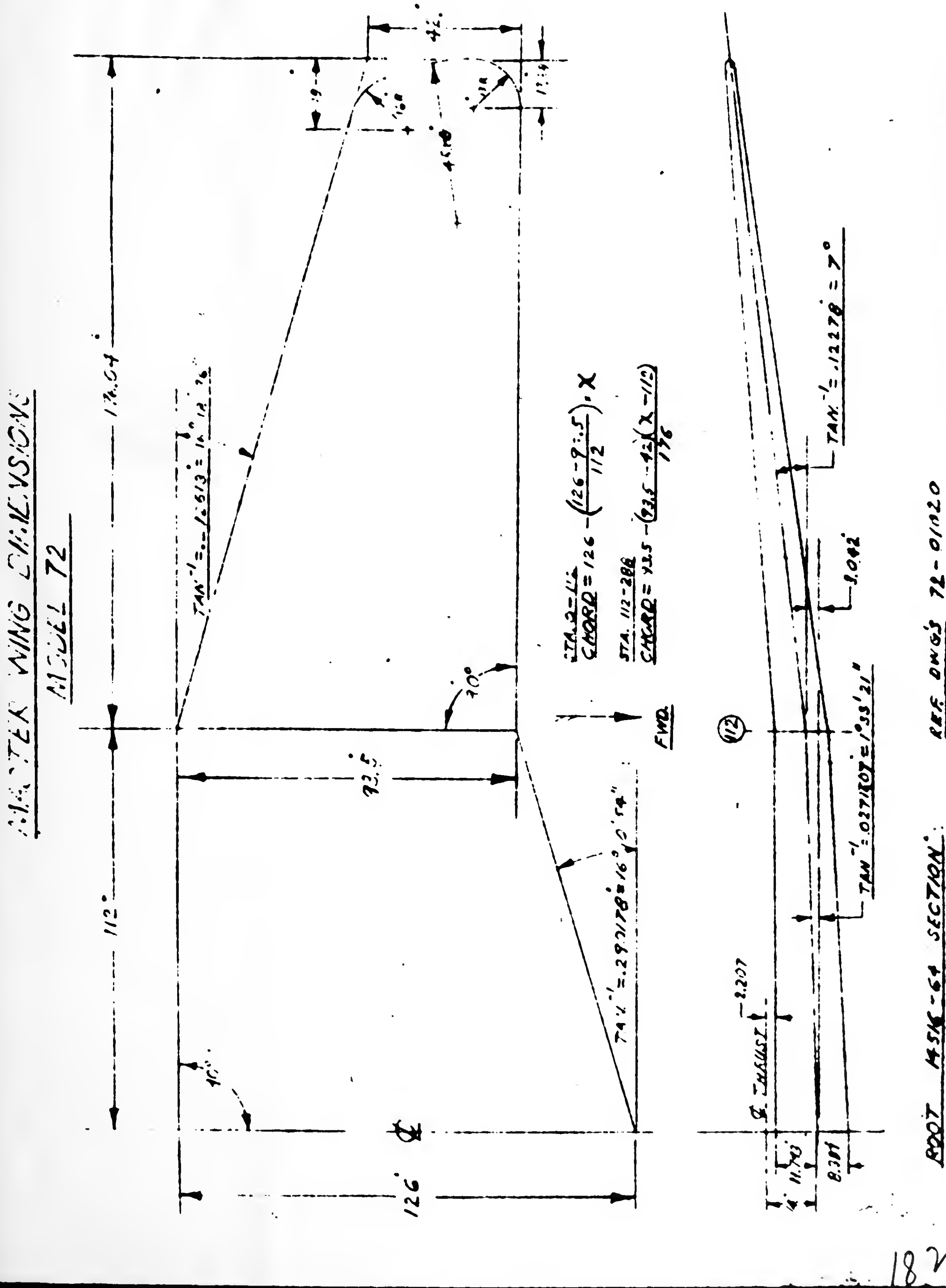
NO.	DATE	BY	CHANGE	PAGES AFFECTED

819

ANALYSIS
PREPARED BY
CHECKED BY
REVISED BY

VULTEE AIRCRAFT
DIVISION AVIATION MANUFACTURING CORP.

PAGE 3 12
REPORT NO. 1722
MODEL 72
DATE 1-25-41



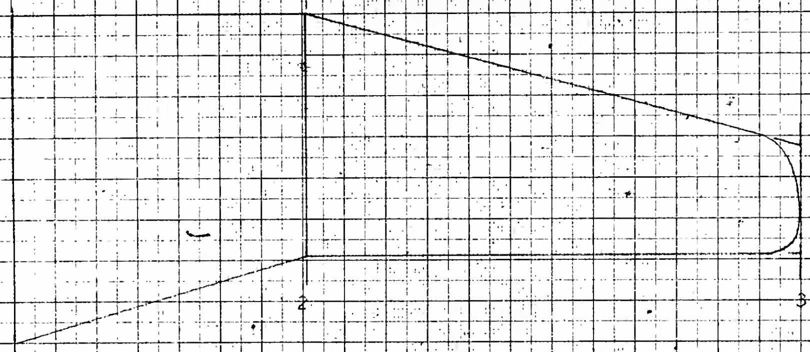
Admitted November 24, 1950.

182

DEFENDANTS' EXHIBIT L

VULTEE MODEL V72 (A35) "VENGEANCE" AIRPLANE

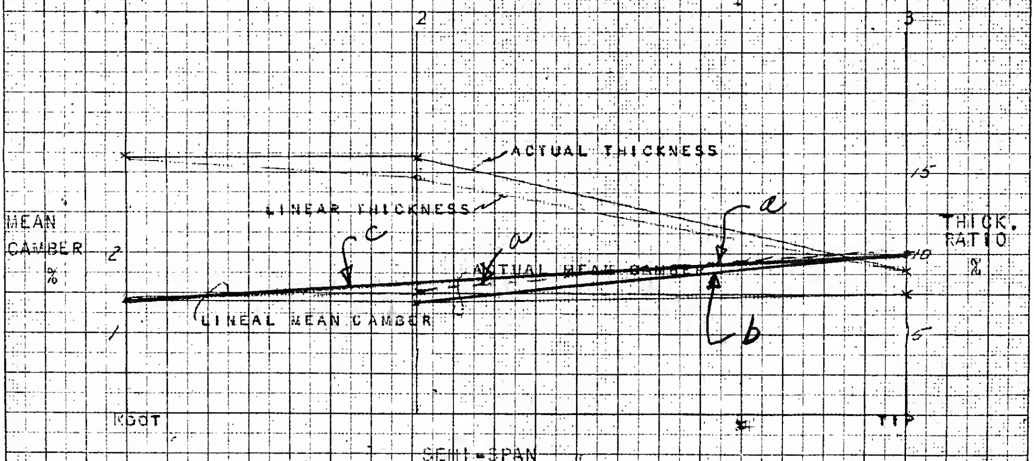
(REFER: DEF. EXH.



SECTION	1	2	3
PROFILE	14516-64	14516-64	20509-64
LOCATION % OF SEMI-SPAN	0	38.8	100
MEAN CAMBER % OF CHORD	1.4	1.4	2.0
THICKNESS RATIO % OF CHORD	16.0	16.0	9.0

LINEAR DISTRIBUTION

MEAN CAMBER % OF CHORD	1.4	1.5	2.0
THICKNESS RATIO % OF CHORD	16.0	14.8	9.0



Admitted November 24, 1950.

DEFENDANTS' EXHIBIT M

Contract No. DA-W 535 ac-46

Contract
(Supplies)
JKR:RC

ANMB Preference A-1-D

War Department
(Department)

The Glenn L. Martin Company
(Contractor)

Contract for 500 B-26B Medium Bombardment Airplanes and Spare Parts Therefor and Data.

Amount, $............

Place: Materiel Division, Air Corps, U. S. Army, Wright Field, Dayton, Ohio.

The Finance Officer, U. S. Army, Wright Field, Dayton, Ohio, is designated as the officer to make payments in accordance with this contract.

The supplies and services to be obtained by this instrument are authorized by, are for the purpose set forth in, and are chargeable to Procurement Authority AC 299 P 111-30 A 0021-13, the available balance of which is sufficient to cover cost of same.

AFP: 171981

Letters: June 4, 1941, and June 6, 1941.

Approval recommended: June 24, 1941, for the Chief of the Air Corps.

/s/ W. F. VOLANDT,
Colonel, Air Corps,
Asst. to Chief Mat. Div.

Defendants' Exhibit M—(Continued)

Approved: Jun 26, 1941. By direction of the Secretary of War under the provisions of Section 1(a) Act of July 2, 1940.

/s/ ROBERT P. PATTERSON,
Under Secretary of War.

In Witness Whereof, the parties hereto have executed this contract as of the day and year first above written.

THE UNITED STATES OF AMERICA,

By /s/ G. V. McPIKE,
Major, A.C., Contracting Officer.

JOHN G. SALSMAN,
Major, A.C., U. S. Army,
Contracting Officer.
(Official title)

Two witnesses:

/s/ HARRY T. ROWLAND,
/s/ W. G. EAGER, JR.

[Seal] THE GLENN L. MARTIN COMPANY,
Contractor,

By /s/ J. T. HARTSON,
Vice Pres.,
Baltimore, Maryland.
(Business address)

I, ..., certify that I am the Secretary of the corporation named as contractor herein; that ...., who signed

Defendants' Exhibit M—(Continued)

this contract on behalf of the contractor, was then .... of said corporation; that said contract was duly signed for and in behalf of said corporation by authority of its governing body, and is within the scope of its corporate powers.

[Corporate Seal.]

I hereby certify that, to the best of my knowledge and belief, based upon observation and inquiry, J. T. Hartson, who signed this contract for the Glenn L. Martin Company, had authority to execute the same, and is the individual who signs similiar contracts on behalf of this corporation with the public generally.

/s/ G. V. McPIKE,
Major, Air Corps,
Contracting Officer.

Admitted November 24, 1950.

DEFENDANTS' EXHIBIT N

Contract No. W 535 ac-31733
(8851)

Contract
(Supplies)
WD:jmn

ANMB Preference A-1-A

Allocation Classification System Symbols: USA 1.00

War Department
(Department)

The Glenn L. Martin Company
(Contractor)

Contract for 900 B-26B1 Medium Bombardment Airplanes, Spare Parts and Data.

Defendants' Exhibit M—(Continued)

Amount, $. . . . .

Place: Army Air Forces, Materiel Center, Wright Field, Dayton, Ohio.

The Finance Officer, U. S. Army, Wright Field, Dayton, Ohio, is designated as the officer to make payments in accordance with this contract. The supplies and services to be obtained by this instrument are authorized by, are for the purpose set forth in, and are chargeable to the Procurement Authorities listed hereon, the available balances of which are sufficient to cover the cost of the same.

AC 2312 P 12-09 A 0705-23...... $...........

AC 2382 P 82-09 A 0705-23...... $...........

AFP: 216841 Class. O1-A AAF Stock No. 0121
O1-K

This Formal Contract supersedes Letter Contract Special Form dated July 25, 1942.

Article 52

Approval.—This contract shall be subject to the written approval of the Secretary of War or such individual as said Secretary may designate and shall not be binding until so approved. The date of such approval shall be deemed to be the true date for the purpose of determining all times of performance.

Article 53

Alterations.—The following changes were made in this contract before it was signed by the parties hereto: Articles 15, 16, 16A, 17 to 53, inclusive, on pages 4a, 4a-1 to 4a-6, inclusive, 4b, 4b-1, 4c, 4d, 4d-1, 4e, 4e (cont'd), 4f to 4q, inclusive, and page 5, added as approved by the Director of the Bureau

Defendants' Exhibit M—(Continued)

of the Budget and/or the Under Secretary of War. Paragraph (d) to Article 19 added on page 4b-1. The designation "(a)" added before the title "Taxes" in Article 29, and paragraphs (b) and (c) added to Article 29 on pages 4d and 4d-1. The words "such date or dates . . . representative" in lines 7, 8 and 9 of Article 19 on page 4b hereof, deleted.

In Witness Whereof, the parties hereto have executed this contract as of the day and year first above written.

THE UNITED STATES OF AMERICA,

By /s/ L. S. ROBINSON,
1st Lt., Air Corps,

JOSEPH E. DERHAM,
Lt. Col., Air Corps,
U. S. Army,
Contracting Officer.
(Official title)

THE GLENN L. MARTIN COMPANY,
Contractor,

By /s/ HARRY T. ROWLAND,
Vice President,
Baltimore, Maryland.
(Business address)

Two witnesses:

/s/ W. G. EAGER, JR.,
/s/ G. C. WILLIAMS.

Defendants' Exhibit M—(Continued)

I, M. G. Shook, certify that I am the Assistant Secretary of the corporation named as contractor herein; that Harry T. Rowland, who signed this contract on behalf of the contractor, was then Vice President of said corporation; that said contract was duly signed for and in behalf of said corporation by authority of its governing body, and is within the scope of its corporate powers.

[Corporate Seal]

/s/ M. G. SHOOK,
Ass't Sec'y

I hereby certify that, to the best of my knowledge and belief, based upon observation and inquiry, ..., who signed this contract for the ...., had authority to execute the same, and is the individual who signs similar contracts on behalf of this corporation with the public generally.

.........................,
Contracting Officer.

W 535 ac-31733

Admitted November 24, 1950.

DEFENDANTS' EXHIBIT O

THE GLENN L. MARTIN CO.
BALTIMORE, MD.

TRADE MARK

PACKING ORDER

Packing Order 7505-90

R-45-14515-B.K.

Filled from accept letter

DATE

No. 1-22492

DA W535 ac-46

YOUR ORDER

OUR ORDER S-0550

B/L No.

CAR No.

TO
Officer, G.F.E.
ield, Dayton, Ohio
DESTINATION UNKNOWN

) PILOT:-

QUAN.	PART NO.	DESCRIPTION	ATTACHING PARTS QUAN.	PART NO.	NAME	WEIGHT

INSPECTED BY THE OFFICE OF THE A.A.F. RESIDENT REPRESENTATIVE
C/O THE GLENN L. MARTIN COMPANY, BALTIMORE, MARYLAND

e 16

R-344000 Airplane, Martin Twin Engine Medium Bombardment
Air Corps Model B-26-B35MA - Martin Model 179

MARTIN NO 3778 A.C. NO. 41-32064 SERIAL NO. FB-909

In accordance with requirements of U.S. Air Corps Spec. C-213 dated January 25, 1939 and Amendment thereto and as amended by The Glenn L. Martin Company Specification #88B1, revised January 23, 1943, including Change Orders and Engineering Releases, as listed on Page Nos. 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 13, 14 and 15 hereof and complete with Government Furnished Equipment.

Airplane completely setup, serviced and ready for flight at our field. Furnished and supplied all fuel, oil and cooling fluid necessary for engine tests, flight tests (to Spec. R-1880-D dated December 1, 1938) and amount required for fly-away as designated by ferry pilot. (Total number of engine fuel not to exceed 300 gallons).

960 GALLONS GASOLINE

"CERTIFIED IN ACCORDANCE WITH CONTRACT DA W535 ac-46, SPECIFICATION, DEVIATIONS AND CHANGE ORDERS PERTAINING THERETO AND TO INCORPORATE ALL ITEMS OF GOVERNMENT MATERIAL LISTED HEREON."

THE GLENN L. MARTIN CO. REPRESENTATIVE

Lawrence J. Kane
Chief Inspector

"I CERTIFY THAT I HAVE EXERCISED DUE DILIGENCE AND HAVE NO REASON TO BELIEVE THAT THE MATERIAL LISTED HEREON HAS NOT BEEN PROPERLY USED AS CERTIFIED BY THE CONTRACTOR."

Ira J. Grey
A. A. F. INSPECTOR, U.S. ARMY

DATE ACCEPTED: 7-17-43

Admitted November 24, 1950.

S TO) ASSEMBLY
CTION-ARMY-NAVY

SHEET NO. 1 OF

ISSUED BY Prod. Dept. DATE June 9, 1943

WAR DEPT. AAF
FORM NO. 263A
APPROVED.........., 1943

Part 6

CONTRACTOR'S DELIVERY RECEIPT

DATE SEP 2 0 1944

AIRPLANE MODEL B26G-10-MA A. A. F. SERIAL NO. 43- 34593 CONTRACT NO. W535 ac-31733 AC Spec. C-215

NAME OF CONTRACTOR .... MFR'S SERIAL NO. 8718 SPEC. NO. GLM Spec. No. 8881 dated January 1, 1943

I hereby certify that this airplane, at the time of delivery to the Ferry Pilot, was equipped with items as listed in preceding parts of this Form 263A, with the exceptions noted on Shortage List, Part 5.

(signed) Russell B...
CONTRACTOR'S INSPECTOR

A. A. F. INSPECTOR-IN-CHARGE
(or authorized agent)

Mat. Comm. Eastern Procurement Dist.
Rank and Organization

(signed) Clement L. Curroy
Lt. Clement L. Curroy
A. A. F. RESIDENT REPRESENTATIVE
(or authorized agent)

Procurement Inspector
Official Designation

1 st. Lt. Air Corps
Rank and Organization

Operations Officer
Official Designation

DATE..........

I hereby certify that, to the best of my knowledge, no items of equipment, as delivered to me by the Contractor, have been removed from this airplane while in my custody, with the following exceptions:

....
(if none, so indicate)

....
SIGNATURE OF FERRY PILOT

Note: The Ferry Pilot is not required to make an actual check of the airplane for completeness.

....
Rank and Organization

....
Official Designation

RECEIPT FOR FERRY PILOT

MODIFICATION CENTER
or Other Station.......... DATE..........

Receipt is hereby acknowledged of Airplane Model..........

A.A.F. Ser. No..........

Delivered this date from..........

Name of Contractor

(Signed)..........
Receiving Officer or Ferry Pilot

Rank and Organization

189

DEFENDANTS' EXHIBIT AA

CONTRACTOR'S COPY
NOT TO BE RETURNED

NA 507
(26 Jan. 1942)

This is your official copy of the contract. The ORIGINAL contract only is signed by the parties. Signatures are not required on the contractor's copy.

SUPPLEMENTARY CONTRACT

FIXED PRICE CONTRACT

(SUPPLIES)

Contract No. a LL 76927

Negotiated Contract

NAVY DEPARTMENT
BUREAU OF SUPPLIES AND ACCOUNTS
(Department)

GLENN L. MARTIN COMPANY
(Contractor)

Contract for SEE INSIDE Amount, $

Place FOB CONTRACTOR'S PLANT BALTIMORE, MARYLAND

THIS CONTRACT, entered into this 11th day of July, 19 42

by the UNITED STATES OF AMERICA, hereinafter called the Government, represented by the contracting officer

executing this contract, and GLENN L. MARTIN COMPANY

a corporation organized and existing under the laws of the State of

a partnership consisting of

an individual trading as

of the city of BALTIMORE, in the State of MARYLAND

hereinafter called the contractor, witnesseth that the parties hereto do mutually agree as follows:

ARTICLE 1. Scope of this Contract.--The contractor shall furnish and deliver all of the supplies or services described in Schedule A attached hereto for the consideration stated opposite each item or each lot in Schedule A, in strict accordance with the specifications, schedules and drawings attached to or designated in Schedule A, all of which are made a part hereof. Deliveries and payment of the contract price shall be made as stated in Schedule A. The rights and obligations of the parties hereto shall be subject to the provisions contained in Article 1 to 11 of this contract, inclusive, as well as the provisions of Schedule A. In the event of any inconsistency between the provisions of the said Articles and the provisions of Schedule A, the latter shall be deemed to control to the extent of such inconsistency.

ARTICLE 2. Changes.--Where the supplies to be furnished are to be specially manufactured in accordance with drawings and specifications, the contracting officer may at any time, by a written order, and without notice to the sureties, make changes in the drawings or specifications, except Federal Specifications. Changes as to shipment and packing of all supplies may also be made as above provided. If such changes cause an increase or decrease in the cost of performing this contract, or in the time required for its performance, an equitable adjustment shall be made and the contract shall be modified in writing accordingly. Any claim for adjustment under this article must be asserted within 10 days from the date the change is ordered: Provided, however, That the contracting officer, if he determines that the facts justify such action, may receive, consider and adjust any such claim asserted at any time prior to the date of final settlement of the contract. If the parties fail to agree upon the adjustment to be made the dispute shall be determined as provided in Article 9 hereof. But nothing provided in this article shall excuse the contractor from proceeding with the contract as changed.

ARTICLE 3. Extras.--Except as otherwise herein provided, no charge for extras will be allowed unless the same have been ordered in writing by the contracting officer and the price stated in such order.

ARTICLE 4. Responsibility for supplies tendered.--The contractor shall be responsible for the articles or materials covered by this contract until they are delivered at the designated point, but the contractor shall bear all risk on rejected articles or materials after notice of rejection. Where final inspection is at point of origin but delivery by contractor is at some other point, the contractor's responsibility shall continue until delivery is accomplished.

~~ARTICLE 5. Increase or decrease.--Unless otherwise specified, any variation in the quantities called for, not exceeding 10 percent, will be accepted as a compliance with the contract, when caused by conditions of loading, shipping, packing, or allowances in manufacturing processes, and payments shall be adjusted accordingly.~~

ARTICLE 6. Additional security.--Should any surety upon the bond for the performance of this contract become unacceptable to the Government, or if any such surety shall fail to furnish reports as to his financial condition from time to time as requested by the Government, the contractor must promptly furnish such additional security as may be required from time to time to protect the interests of the Government and of persons supplying labor or materials in the prosecution of the work contemplated by the contract.

ARTICLE 7. Officials not to benefit.--No Member of or Delegate to Congress or Resident Commissioner shall be admitted to any share or part of this contract or to any benefit that may arise therefrom, but this provision shall not be construed to extend to this contract if made with a corporation for its general benefit.

ARTICLE 8. Covenant against contingent fees.--The contractor warrants that he has not employed any person to solicit or secure this contract upon any agreement for a commission, percentage, brokerage, or contingent fee. Breach of this warranty shall give the Government the right to annul the contract, or, in its discretion, to deduct from the contract price or consideration the amount of such commission, percentage, brokerage, or contingent fees. This warranty shall not apply to commissions payable by contractors upon contracts or sales secured or made through bona fide established commercial or selling agencies maintained by the contractor for the purpose of securing business.

ARTICLE 9. Disputes.--Except as otherwise specifically provided in this contract, all disputes concerning questions of fact arising under this contract shall be decided by the contracting officer, subject to written appeal by the contractor within 30 days to the head of the department concerned or his duly authorized representative, whose decision shall be final and conclusive upon the parties hereto. In the meantime the contractor shall diligently proceed with performance.

ARTICLE 10. Race Discrimination.--The contractor, in performing the work required by this contract, shall not discriminate against any worker because of race, creed, color, or national origin. The contractor further agrees that each subcontract made under this contract will contain a similar provision with respect to non-discrimination.

ENTERED

APR 7 1943

any assistant head of the executive department involved, and the term "his duly authorized representative" shall mean any person authorized to act for him other than the contracting officer.

(b) The term "contracting officer" as used herein shall include the chief of the Bureau of Supplies and Accounts, the Purchasing Officers in such Bureau, and their duly appointed successors and duly authorized representatives.

IN WITNESS WHEREOF, the parties hereto have executed this contract as of the day and year first above written.

THE UNITED STATES OF AMERICA

By ______________________________

Purchasing Officer,
Bureau of Supplies and Accounts,
Navy Department.
(Official title)

Two witnesses:

V. Seth

G. C. Williams

THE GLENN L MARTIN COMPANY

Contractor.

Harry T. Rowland

Harry T. Rowland, Vice President

Baltimore, Maryland
(Business address)

I, M. G. Shook, certify that I am the Asst. Secretary of the corporation named as contractor herein; that Mr. Rowland who signed this contract on behalf of the contractor, was then Vice Pres. of said corporation; that said contract was duly signed for and in behalf of said corporation by authority of its governing body, and is within the scope of its corporate powers.

M. G. Shook
M. G. Shook

Corporate Seal

I hereby certify that, to the best of my knowledge and belief, based upon observation and inquiry, ______, who signed this contract for the ______, had authority to execute the same, and is the individual who signs similar contracts on behalf of this corporation with the public generally.

____
Contracting Officer.

U.S. Standard Form No. 25 (Revised)
Approved by the Secretary
of the Treasury
Sept. 16, 1935

NO BOND OR OTHER SECURITY REQUIRED

PERFORMANCE BOND
(Construction or Supply)
(As modified for use by the Navy Department)

KNOW ALL MEN BY THESE PRESENTS, That we,

(See Instructions 4, 5 and 6)

as PRINCIPAL and ______ as SURETY,

(See Instructions 2, 3, 4 and 7)

are held and firmly bound unto the United States of America, hereinafter called the Government, in the penal sum of ______ dollars lawful money of the United States, for the payment of which sum well and truly to be made, we bind ourselves, our heirs, executors, administrators, and successors, jointly and severally, firmly by these presents.

THE CONDITION OF THIS OBLIGATION IS SUCH, That whereas the principal entered into a certain contract, hereto attached, with the Government, dated ______, 19__

NOW, THEREFORE, If the principal shall well and truly perform and fulfill all the undertakings, covenants, terms, conditions, and agreements of said contract during the original term of said contract and any extensions thereof that may be granted by the Government, with or without notice to the surety, and during the life of any guaranty required under the contract, and shall also well and truly perform and fulfill all the undertakings, covenants, terms, conditions, and agreements of any and all duly authorized modifications of said contract that may hereafter be made, notice of which modifications to the surety being hereby waived, then, this obligation to be void; otherwise to remain in full force and virtue.

IN WITNESS WHEREOF, the above-bounden parties have executed this instrument under their several seals this ______ day of ______, 19__, the name and corporate seal of each corporate party being hereto affixed and these presents duly signed by its undersigned representative, pursuant to authority of its governing body.

In presence of--

______________________ ______________________ SEAL
(Principal)

______________________ ______________________ SEAL
(Principal)

______________________ ______________________ SEAL
(Surety)

______________________ ______________________ SEAL
(Surety)

The rate of premium on this bond is ______________ per thousand.

Total amount of premium charged, $______________

(The above must be filled in by corporate surety)

If individual sureties sign the above bond the Affidavits and Certificates on the Appended Sheet must be executed.
837P

1 2 6

Admitted November 24, 1950.

DEFENDANTS' EXHIBIT BB

4/11/42

BALTIMORE, MD.

PACKING ORDER

Order Number 5403-93

No. 44001

YOUR ORDER C.76927

OUR ORDER S-1910

B/L No.

CAR No

TO Accepted by I.N.A. Baltimore, USN
For Delivery to: Supply Officer,
Naval Air Station, Norfolk, Virginia

AIR TO DESTINATION
(PILOT -)

QUAN.	PART No.	DESCRIPTION	ATTACHING PARTS QUAN.	PART No.	NAME	WEIGHT
		INSPECTED BY I.N.A. BALTIMORE				
1	1628100	Airplane, Class VPB, Model PBM-3, Martin No. 2908, Navy No. 6455				
		Constructed in accordance with The Glenn L. Martin Company's Detail Specification No. SD-250-3-1A, and requirements of Bureau of Aeronautics Specification SD-250-3-1 dated 6 June 1940, and revisions thereto, complete with Government Furnished Equipment.				
		Airplane completely set up, ground tested and serviced with 1,000 gallons of gasoline (100 Octane) and 80 gallons of oil ready for flight at the Contractor's plant, after acceptance of airplane.				
		See Page No. 2 for record of Navy Changes pretaining to this airplane.				

Due to delivery of airplane being made by Contractor to a Naval Activity at Norfolk, Va. the signature [illegible] is a verification of material listed above and is not a signature of acceptance.

Cleanliness & finish not according to delivery standards.

ALL STRAINERS CLEANED IN THIS AIRPLANE PRIOR TO DELIVERY.

THE SELF-SEALING FUEL CELLS IN THIS AIRPLANE HAVE NOT BEEN SLOSHED FOR USE WITH AROMATIC FUELS - TO BE TAKEN CARE OF LATER.

SIGNATURE [signature] THE GLENN L. MARTIN CO. INSPECTOR

[signature] BALTIMORE INSPECTOR OF NAVAL AIRCRAFT

192

TO ASSEMBLY - ARMY-NAVY SHEET NO. 1 OF 3 ISSUED BY DATE

16,115-43

THE GLENN L. MARTIN CO.

BALTIMORE, MD.

MARTIN USA — TRADE MARK

PACKING ORDER

Packing Order No. 5403-93

DATE JAN 12 1944

No. 1-81105

YOUR ORDER Suppl. Contr. NOs-L176927

OUR ORDER GLM Suppl. No. 136D 5400 Series

B/L No.

CAR No.

...d by: I.N.A. Baltimore, USN

...y-Away Delivery R 44 37896

...to Destination (Pilot:

QUAN.	PART NO.	DESCRIPTION	ATTACHING PARTS QUAN.	PART NO.	NAME	WEIGHT
	13186	INSPECTED BY I.N.A. BALTIMORE				
1	162D100	Airplane, Class VPB, Model PBM-3D				
		MARTIN NO. 7990 NAVY NO. 48219				
		This airplane furnished, completely assembled and ready for flight in accordance with The Glenn L. Martin Company Specification SD-250-3-1A dated June 26, 1941, as modified by the changes listed in Exhibit A of Contract (such Specification as so modified being herein after called Specification SD-250-3-1A) and complete with Government Furnished Equipment.				
17	1944	Airplane serviced with 700 gallons of gasoline (100 Octane), and 30 gallons of oil.				
		Auxiliary Bomb Bay Fuel Tanks to be forwarded under separated Notice of Shipment.				
		ALL STRAINERS CLEANED IN THIS AIRPLANE PRIOR TO DELIVERY				

SIGNATURES: J Dealon McDaniel — THE GLENN L. MARTIN CO. INSPECTOR

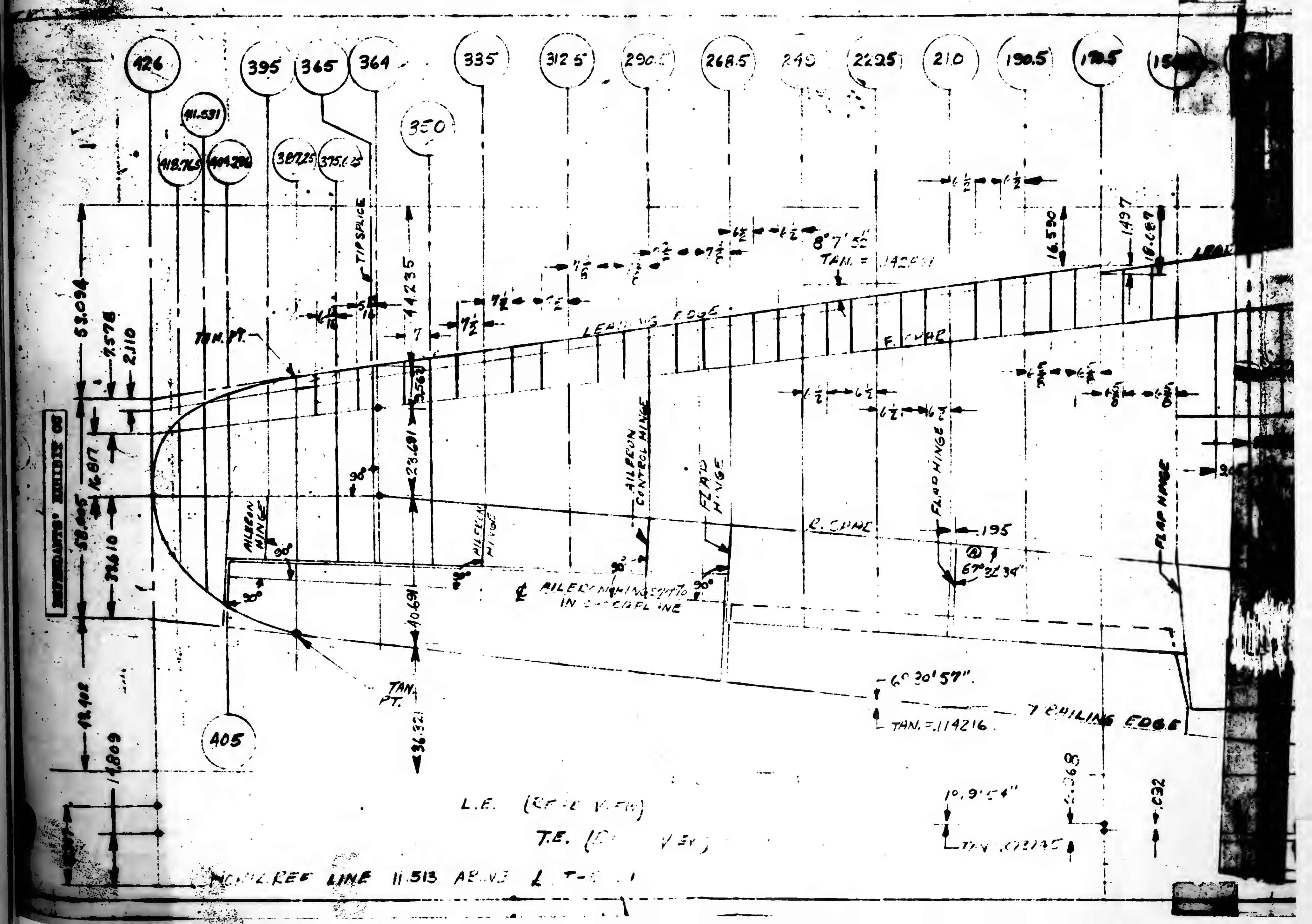
Victor E. Sage, by direction — BALTIMORE INSPECTOR OF NAVAL AIRCRAFT

ALL changes, additions and deletions are void unless signed by both GLMCO and INA.

DATE ACCEPTED: 1/7/44 1010

Admitted November 24, 1950.

...TO) ASSEMBLY-...ION-ARMY-NAVY SHEET NO. 1 OF ____ ISSUED BY Prod. ... DATE



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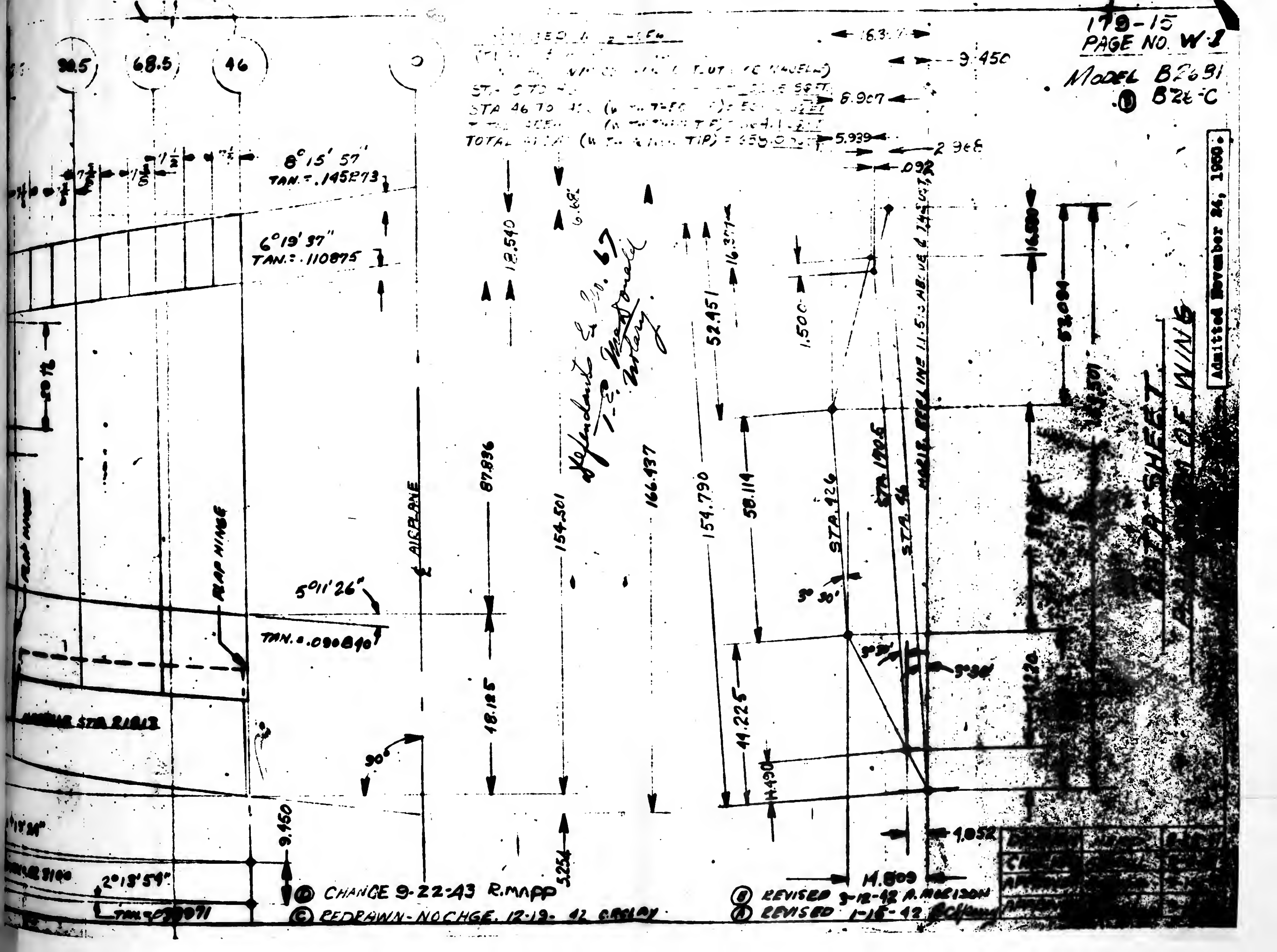
387.25
375.625
350
TIPSPLICE
44.235
53.094
7.578
2.110
TAN. PT.
LEADING EDGE
8°7'52"
16.590
1.497
18.087
F. SPAR
7
5.562
23.691
AILERON CONTROL HINGE
FLAP HINGE
FLAP HINGE
FLAP HINGE
AILERON HINGE
AILERON HINGE
.195
67°32'34"
90°
16.817
58.005
32.610
40.691
TAN. PT.
36.321
405
6°20'57"
TAN.=.114216
TRAILING EDGE
12.402
14.809
DEFENDANTS' EXHIBIT 68
L.E.
T.E.
1°,9'54"
.032
HORIZ. REF LINE 11.513 ABOVE


179-15
PAGE NO. W-8
MODEL B26-B1
B26-C
Admitted November 26, 1950.
TOTAL
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TAN. = .145273
6° 19' 37"
TAN. = .110875
5° 11' 26"
TAN. = .090890
2° 13' 59"
90°
3° 30'
68.5
46
12.540
87.836
154.501
166.437
154.790
52.451
16.377
58.114
44.225
1.500
1.490
18.125
9.450
5.254
16.830
4.809
1.052
8.907
5.939
2.968
.092
9.450
FLAP HINGE
AIRPLANE
STA. 126
STA. 56
Defendants Ex. No. 67
① CHANGE 9-22-43 R. MAPP
Ⓒ REDRAWN-NO CHGE. 12-19-42
Ⓑ REVISED
Ⓐ REVISED 1-15-42

No. 12885

United States Court of Appeals

for the Ninth Circuit.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION and AMERICAN AIR LINES, INC.,

Appellants,

vs.

MAURICE A. GARBELL, INC., and GARBELL RESEARCH FOUNDATION,

Appellees.

Transcript of Record

Volume IV

Book of Exhibits

(Pages 835 to 1005)

Appeal from the United States District Court, Southern District of California, Central Division.

Phillips & Van Orden Co., 870 Brannan Street, San Francisco, Calif.

DEFENDANTS' EXHIBIT FF

Engineering Report
Date: November, 1941

No. 1484, Vol. I
No. Pages, 185

The Glenn L. Martin Company
Baltimore

Model B-26 B1 & C

Detail Specification GLM Spec. #88B Contract
DA-W535AC-46
DA-W535AC-19342

Stress Analysis
of Wing

Prepared By:

/s/ VINCENT COUDELLO,
/s/ PETER N. LAYTON, III,
/s/ F. LEIGH NOYES.

Checked By:

/s/ RICHARD K. WENTZ,
/s/ C. H. RIS,
/s/ LEON R. COBAUGH.

Approved By:

/s/ P. C. MEDINA,
A Project Stress Engineer,
/s/ G. N. MANGURIAN,
A Structural Design Engr.,
/s/ G. L. BRYAN, JR.,
Chief Structural Engr.

Revisions

Date	Pages Affected	By	Remarks
. . . .	. . . .	. . . .	. . . .

Defendants' Exhibit FF—(Continued)

Analysis of Wing

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(a) U. S. Army Air Corps "Spec. No. X-1803-A, Stress Analysis Criteria," dated Nov. 15, 1938.

(b) G.L.M. Spec. No. 88; "Detail Specification for Air Corps, Model B-26 Bombardment Airplane, Twin Engine."

(c) Army - Navy - Commerce Bulletin, ANC-1(1); April, 1938, "Spanwise Air Load Distribution."

(d) N.A.C.A. Confidential Memo. Report of Oct. 3, 1939, "Wing Tunnel Tests of a 1/8 Scale Model of Martin 179 Bomber."

(e) N.A.C.A. Confidential Memo. Report of Oct. 7, 1939, "Additional Tests of 1/8 Scale Model of Martin 179 Bomber."

(f) G.L.M. Engineering Report No. 1483, "Stress Analysis of Basic Flight Criteria, Model B-26, B1 & C."

(g) G.L.M. Model B-26, B1 & C Data Book.

(h) G.L.M. Engineering Report No. 1499, "Weight and Balance Report, Model B-26, B1 & C."

(i) U. S. Army Air Corps "Handbook of Instructions for Airplane Designers," 8th Edition, revised to July 1, 1939.

(j) Letter to G. L. Martin Co. from U. S. Army Material Division, CKM-rf-51, October 28, 1939.

Defendants' Exhibit FF—(Continued)

(k) G.L.M. Engineering Report No. 1486, "Stress Analysis of Landing Gear, Model B-26, B1 & C."

(l) G.L.M. Engineering Report No. 1485, "Stress Analysis of Fuselage, Model B-26, B1 & C."

(m) G.L.M. Engineering Report No. 1154, "Stress Analysis of Wing, Model B-26."

(n) Army-Navy-Civil Bulletin, ANC-1(2), "Chordwise Airload Distribution"—Feb., 1939.

Introduction

The stress analysis of the wing for Air Corps Bomber Model B-26-B1&C (Martin Model 179-15) consists of three volumes. Volume I contains the computations of the basic design loads, Volume II contains the stress analysis of the wing box, and Volume III contains the stress analysis of the ribs and structural details. The analysis of the fittings is made in G.L.M. Engineering Report No. 1488.

Volume I Consists of Five Parts:

Part 1—General Data.

Part 2—Spanwise and Chordwise Air Load Distribution.

Part 3—Air Load Computations—Shears, Bending Moments, and Torsional Moments.

Part 4—Unit Load Computations—Dead Weight, Shears, Bending Moments, and Torsional Moments.

Part 5—Net Design Load Computations—Shears, Bending Moments, and Torsional Moments.

Defendants' Exhibit FF—(Continued)

Part 1

General Data

Wing Geometry—Reference (f) and (g)

Span .... 71 ft.

Area .... 659 sq. ft.

Airfoil Section

Station 46.........N.A.C.A. 0016.7-64

Tip (theoretical).....Martin Revision

Root Chord (theoretical at CL).. 166.75 inches

Tip Chord (theoretical) .... 58.12 inches

Incidence (relative to thrust line)....+ $3\frac{1}{2}°$

Mean Aerodynamic Chord...121.5 in. (ref. (f))

Weights

Normal Gross Weight (Ref. (f))....31,000 lbs.

Minimum Flying Weight (Ref. (f)).24,200 lbs.

Overload Gross Weight (Ref. (f))...35,500 lbs. (Max. Range)

Sign Conventions

The planes of the wing spar webs are perpendicular to a horizontal plane through the thrust line. Forces are resolved into components parallel and normal to the thrust line. Loads and accelerations referred to as being in the "beam" direction are normal to the thrust line while those in the "chord" direction are parallel to it.

Loads and accelerations are positive when up, aft, and out.

Positive beam bending moment causes compression in the upper surface of the wing.

Defendants' Exhibit FF—(Continued)

Positive chord bending moment causes compression in the rear spar.

Positive torsional moments tend to stall the airplane.

All dimensions of lengths and areas are in inches and square inches, respectively, unless otherwise noted.

Reference Axis

A reference axis is used for the calculation of torsional moments. This axis is the intersection line of a horizontal plane through the thrust line and a plane normal to the thrust line which passes through the leading edge of the root chord. (See pages 7 and 75.)

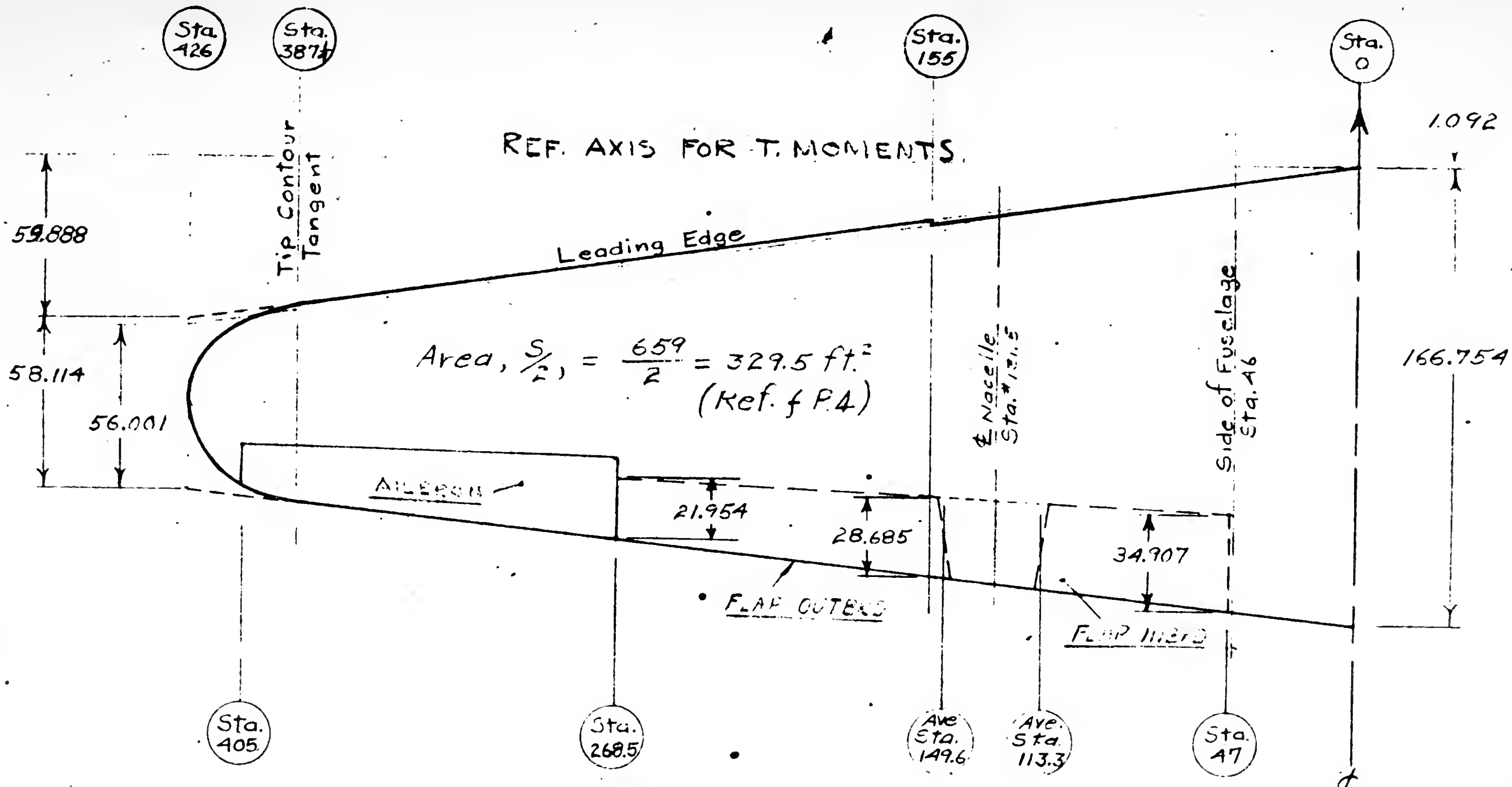
In the detailed analysis of any section of the wing, the torsional moments are transferred from this axis to the elastic axis of the wing section under consideration.

Aerodynamic Center (a.c.)

Aerodynamic loads are assumed to be concentrated at the aerodynamic center. Although the a.c. location along the span does not actually vary linearly, the slight discrepancy introduced by assuming it so is negligible. Therefor for convenience in calculating the torsional moments, a line of aerodynamic centers is assumed, which varies from 23% C at Station 46 to 24% C at theoretical tip. (Ref. page 75.)

Fig. # 1 Diagramatic Sketch of Wing Ref. (9), F. VI-1

(In True Chord Plane)



"True" Chords (Inboard of Sta. # 387¼)

Inboard of Sta. # 155:

$$C = 166.754 - \left(\frac{166.754 - 56.001}{426}\right) \times Sta.$$

$$= 166.764 - .25998(Sta.)$$

Outboard of Sta. # 155:

$$C = 167.846 - \left(\frac{167.846 - 58.114}{426}\right) Sta.$$

$$= 167.846 - .25759(Sta.)$$

83

PART No. 2

Span-wise Distribution of Wing Coefficients

The span-wise distribution of wing coefficients is obtained for two conditions:- Wing with flaps neutral (page 9 to 28) and wing with flaps deflected 45° (page 29 to 43).

Since the wing has an effective twist (drooped nose and modified trailing edge) outboard of station 155, the "general method" of Ref. (c) is used to obtain the span distribution of lift and drag coefficients. The distribution of a twisted wing requires two steps, the basic and the additional lift distribution.

The wing tapers uniformly in thickness from tip to root. The chord tapers from theoretical tip to station 155. The chord inboard of 155 is slightly reduced and is assumed to taper uniformly to ℄ airplane (see page 7).

The aerodynamic characteristics of the wing are determined from those of the airfoil sections between station 46 (NACA 0016.7-64) and the theoretical tip (NACA 0010-64 with dropped nose and modified trailing edge). The data obtained from these airfoils are correlated with the characteristics of a similarly shaped wing tested in the wind tunnel.

Wing with Flap Neutral

The basic $\frac{dC_L}{d\alpha}$ vs span (page 10) is adjusted in order to obtain the corrected slope of .072 for the actual A.R. of 7.65 (see page 15).

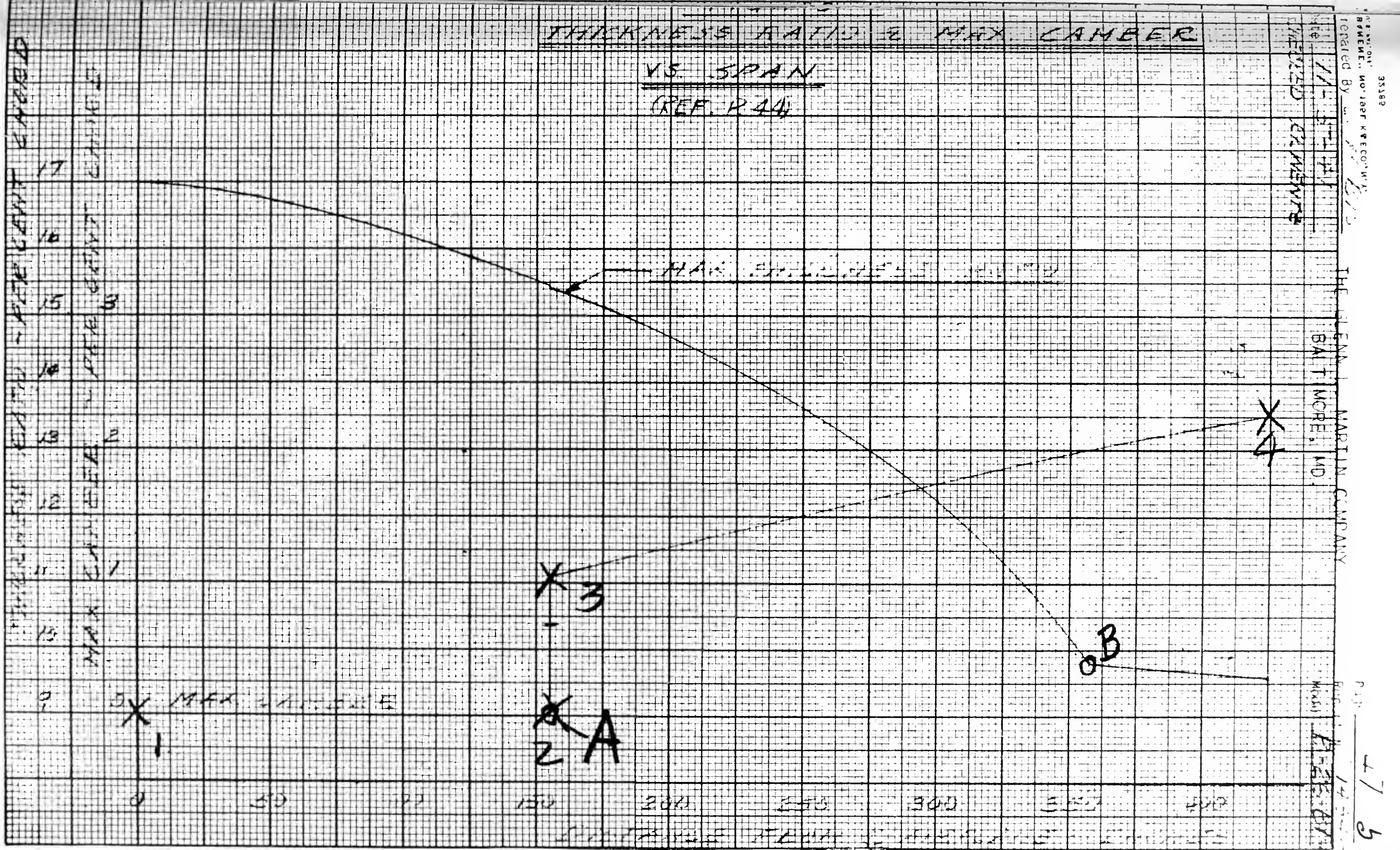
For the "basic lift distribution" the absolute angles of incidence are estimated (as shown on page 9) to determine the lift distribution which depends on the effective twist and is the same for all angles of attack.

The total lift distributions corresponding to the critical design flight conditions are determined by adding the basic and the additional distributions as shown on page 23 and figure 4 , page 24).

The variation of C_{D_0} is adjusted (fig. 5 p. 26) to give the average C_{D_0} = .0085 (Aerodynamic estimate) over the entire wing.

The C_{d_0} is assumed to have the same variation along the span as C_{d_i} (see page 25).

The total wing drag distribution for the critical design conditions is shown on page 27 and plotted on figure 6 , page 28.



Admitted November 24, 1950.

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Admitted November 24, 1950.

849

Defendants' Exhibit I I

SUMMARY OF CHANGES IN THE WING GEOMETRY

OF

THE PBM-3

Engineering Report No. 1339

The Glenn L. Martin Company
Baltimore, Maryland
August 16, 1940

Prepared by: D. Goddard

Checked by:

Approved by: Vernon Outman
Chief of Aerodynamics

Approved by:
Chief Research Engineer

INTRODUCTION

Certain changes have been made in the wing geometry of the PBM-3 airplane as compared to the wing of the PBM-1 airplane.

The changes listed below are discussed in the following pages indicating why the changes were made and the improvement to be achieved by each.

The changes are as follows:

1. The wing has been swept back.
2. The thickness of the wing has been increased.
3. The tip plan form has been modified.
4. The form of the leading edge forward of the spar has been changed outboard of the gull.
5. The dihedral of the outer panel has been reduced.
6. The span of the gull portion of the wing has been increased.
7. The wing taper is straight from the ship ℄ to the wing tip.

The changes are discussed individually in the following pages.

DISCUSSION OF THE CHANGES

1 - WING SWEEP-BACK

The theoretical tip chord of the PBM-3 wing has been swept back by an amount which provides a margin of 4% between the maximum rearward c.g. location in percent of the M.A.C. and the c.g. location for which the static longitudinal stability is neutral. This neutral point is at 34.6% and the most aft c.g. is at about 31.1%. Hence the prescribed sweepback gives satisfactory balance and longitudinal stability.

The wing geometry for the PBM-3 is shown in Figure 1 and the geometry for the PBM-1 is shown in Figure 2.

2 - WING THICKNESS DISTRIBUTION

The wing thickness tapers linearly from the ℄ of the ship to the theoretical wing tip. The section at the ℄ is the 23020 and, at the tip, a modified 23010. The PBM-1 wing was 23020 at the ℄ to modified 23006 at the tip.

The above change in thickness was made to provide greater structural stiffness and to improve the stall characteristics toward the wing tip through use of a thicker section which increases the section C_L maximum.

A comparison of the thickness distribution for PBM-3 and PBM-1 is shown in Figure 3.

The increase in wing thickness causes an estimated 0.5 mph top speed decrease.

3 - TIP PLAN FORM

The tip plan form has been modified from the previous Army tip used on the PBM-1 for reasons of appearance.

4 - OUTER WING LEADING EDGE

The nose section contour forward of the spar has been changed to the form shown in Figure 4. This nose section at station 668 is faired linearly into the 23019.024 section at the gull. (Station 173.5)

4 - OUTER WING LEADING EDGE - Contd.

The purpose of this change is to increase the local C_L maximum toward the tip by increasing the camber of the airfoil and moving the maximum camber forward on the cord. This change also tends to delay the angle of attack at which the tip section will stall. The nose radius of the outer wing section has been increased appreciably by this change as shown in Figure 5, where a 4410 tip has been compared with the PBM-1 and the PBM-3 nose radius variation with span. The combined effect of the blunt nose and camber increase is to produce a flat-top lift curve by moving the transition point aft on the wing surface.

Figure 6 gives a comparison of the camber distribution along the span for the PBM-1 and PBM-3 and for the same wing with a 4410 tip.

The effect of the so-called drooped-nose (Figure 4) on the total airplane drag has been estimated from wind tunnel test on a medium bomber with the same type nose section. The drag polar for this model with and without the droop-nose is shown in Figure 7. ΔC_{D_p} = .0002 at C_L = .35. Since the droop-nose covers about 75% of the span of the PBM-3 and about 38% of the span of the medium bomber, the drag increment for the PBM-3 is estimated at ΔC_{D_p} = .0004. The corresponding decrease in top speed is one mph. (1)

5 - WING DIHEDRAL

The dihedral of the top skin of the outer wing in the chord plane has been made 0° at the 30% chord stations. This was done in order to reduce the rate of change of rolling moment coefficient with angle of yaw, $\frac{dC_l}{d\psi}$, as much as possible and yet not give the wing a drooped appearance. Reducing the value of $\frac{dC_l}{d\psi}$ tends to reduce the possibility of the occurrence of a Dutch Roll condition. The combination

5 - WING DIHEDRAL - Contd.

of reduced dihedral and increased vertical tail area will materially aid this situation.

6 - GULL SPAN

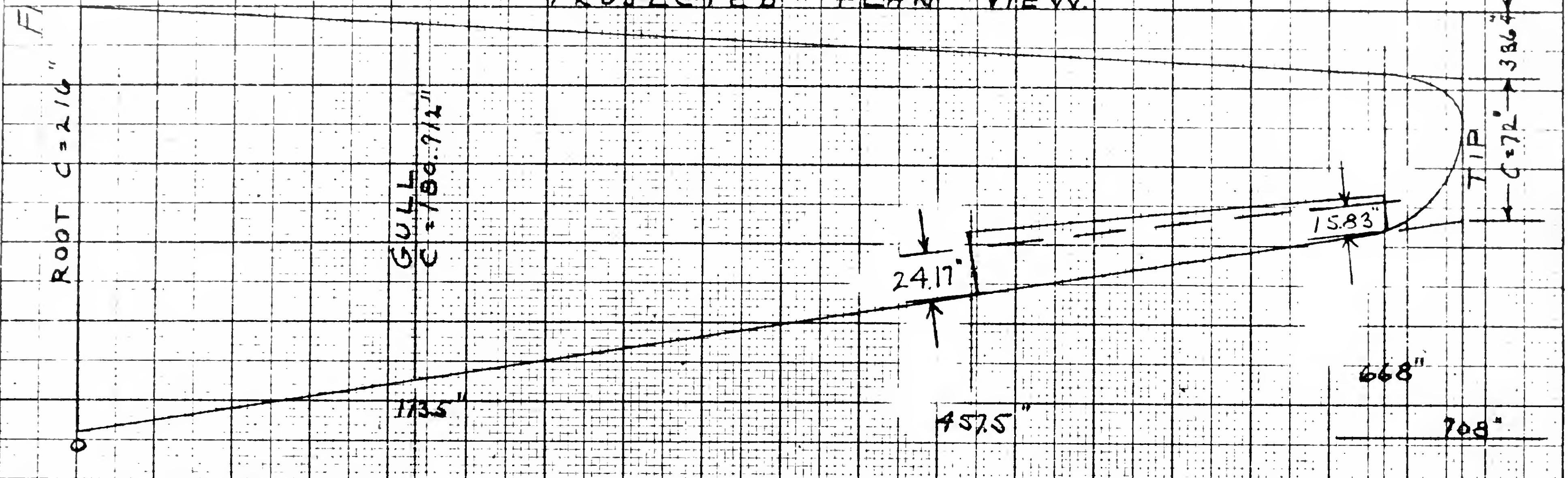
The span of the inner wing (the gull) has been increased twelve inches on either side of the airplane ℄ in order to make room for the nacelle bomb bay, which holds 4-1000 lb. bombs, and still maintain the same spanwise location of the nacelle ℄ as was the case for the PBM-1.

7 - WING PLAN-FORM TAPER

The PBM-3 plan form taper is maintained straight from the ship ℄ to the tip, just as was done on PBM-1.

FIG. 1

162-B WING
PROJECTED PLAN VIEW.



23010 TIP: (DROOPED NOSE)

43.2"	34.3"	THICKNESS - IN.	9.1"	7.2"
20	19.024	% THICKNESS	11.5	10
4.15	3.80	NOSE RADIUS - % CHORD	2.56	1.10
10.00	10.53	MAX. MEANLINE CAMBER - % THICKNESS	30.80	20.0

4410 TIP.

43.2"	34.3"	THICKNESS - IN.	9.1"	7.2"
20	19.024	% THICKNESS	11.5	10
4.15	3.80	NOSE RADIUS - % CHORD	1.45	1.10
10.00	10.53	MAX. MEANLINE CAMBER - % THICKNESS	32.10	40.0

SCALE 1:80

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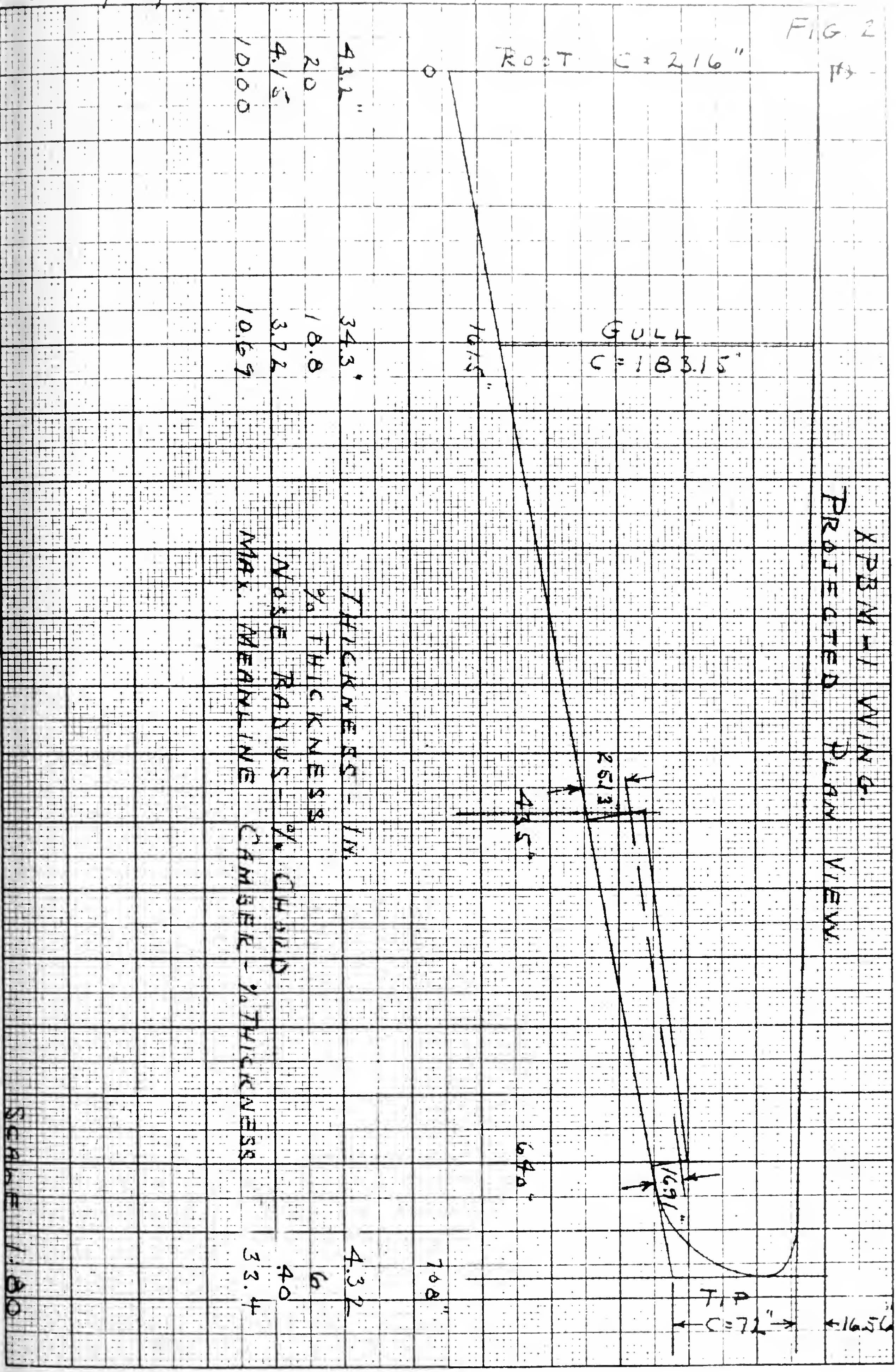
Date 7/31/40 BALTIMORE, MD.



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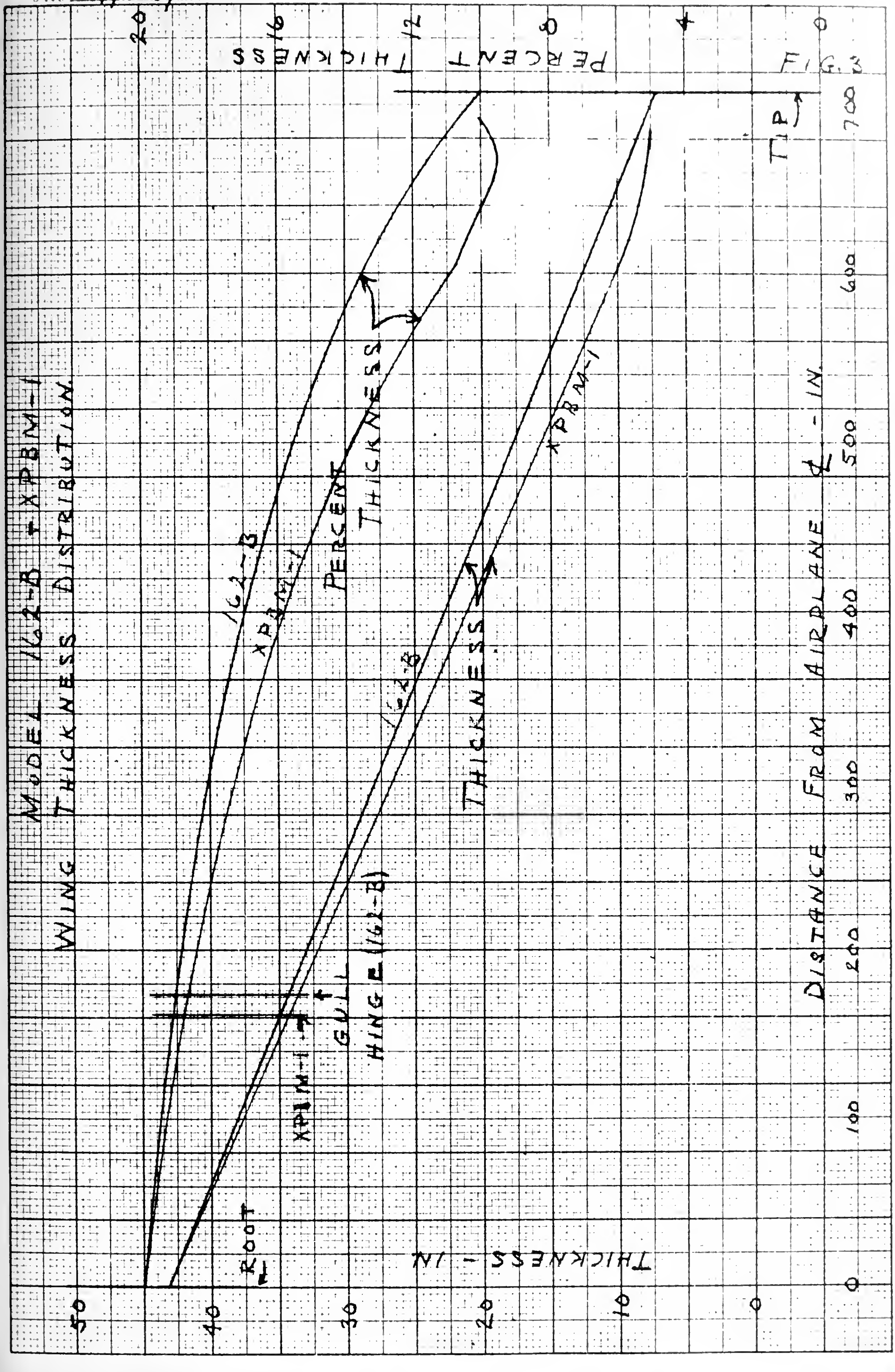
Date 7/23/40

THE GLENN L. MARTIN COMPANY
BALTIMORE, MD.

856

Report No. 1337
Model 162-B

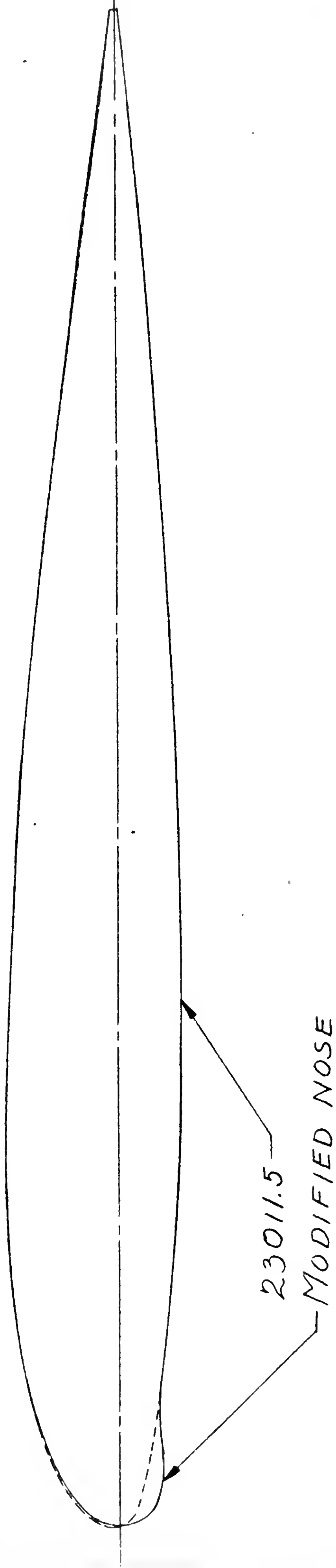


GLM FORM 7..

LENN L. MARTIN COMPANY
MORE, MARYLAND

MODEL 857
PAGE No.
FIG. 4

MARTIN MODEL XPBM-3
WING SECTION AT 668



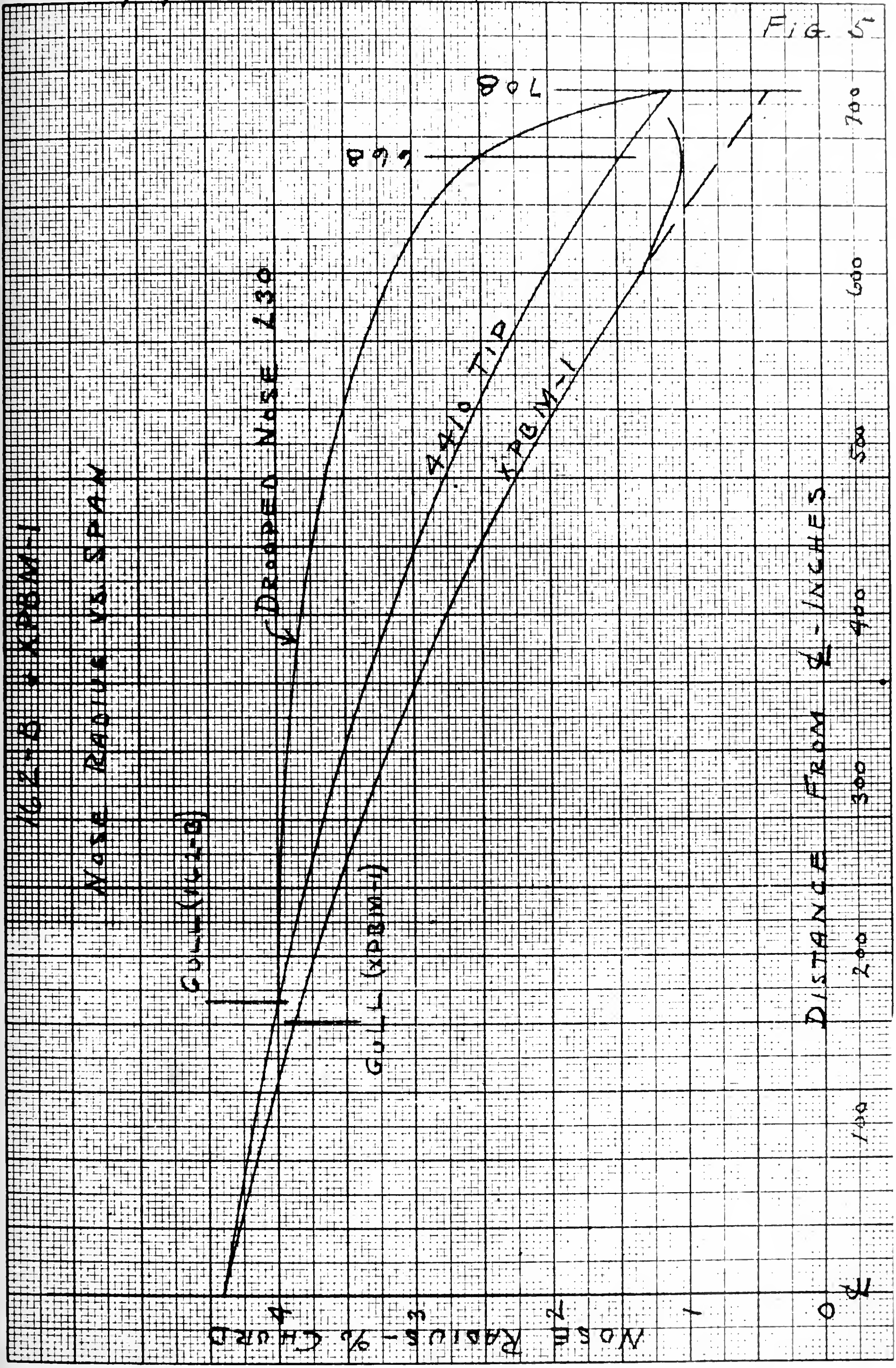
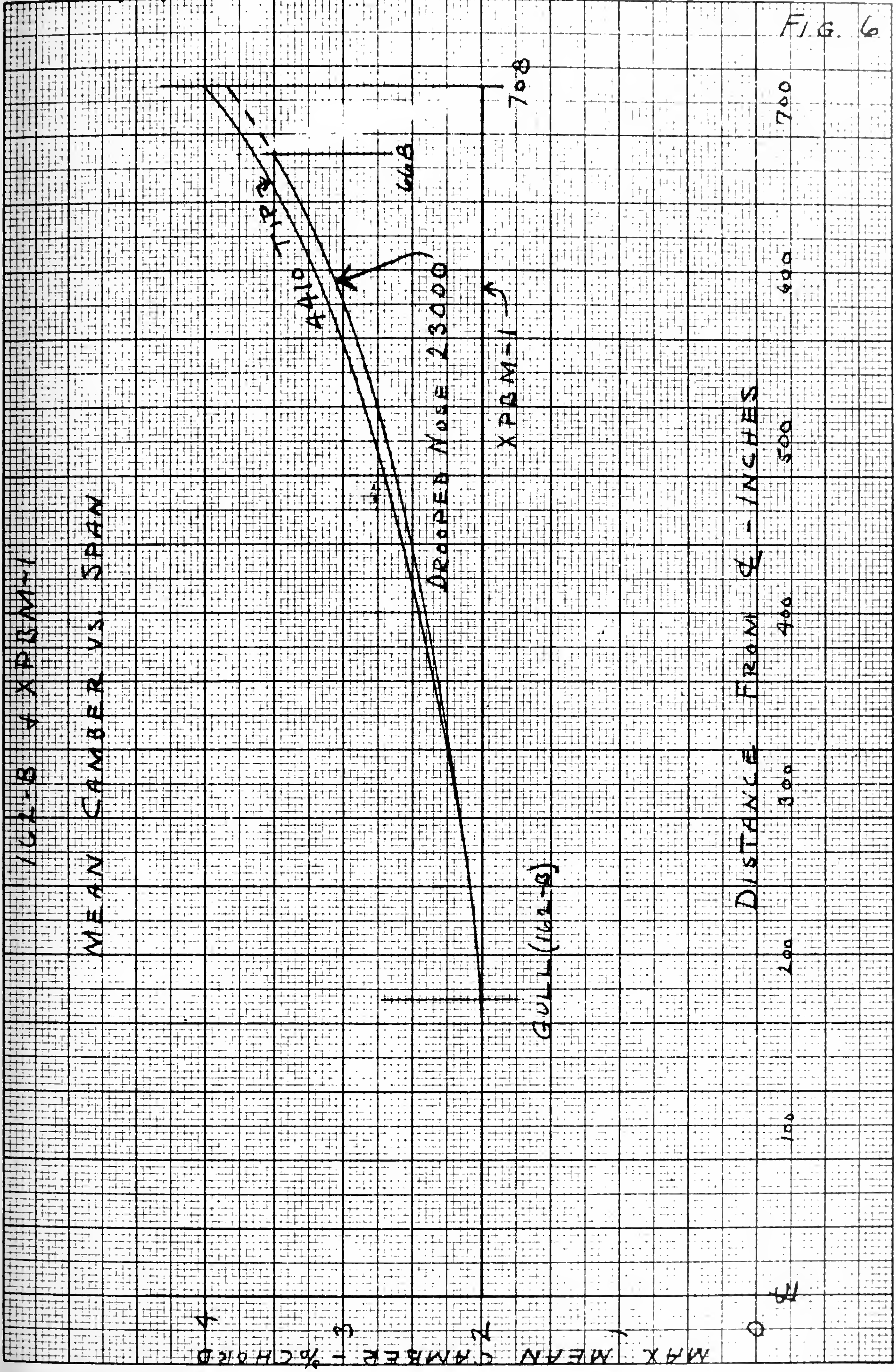


FIG. 5
162-B & XPBM-1
NOSE RADIUS VS. SPAN
DROOPED NOSE L30
4410 TIP
XPBM-1
GULL (162-B)
GULL (XPBM-1)
DISTANCE FROM ℄ - INCHES
0
100
200
300
400
500
600
700
NOSE RADIUS - % CHORD
1
2
3
4

Prepared By [signature] THE GLENN L. MARTIN COMPANY
BALTIMORE, MD.

Date 8/1/40

FIG. 6



GLM FORM 733

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BALTIMORE, MD.

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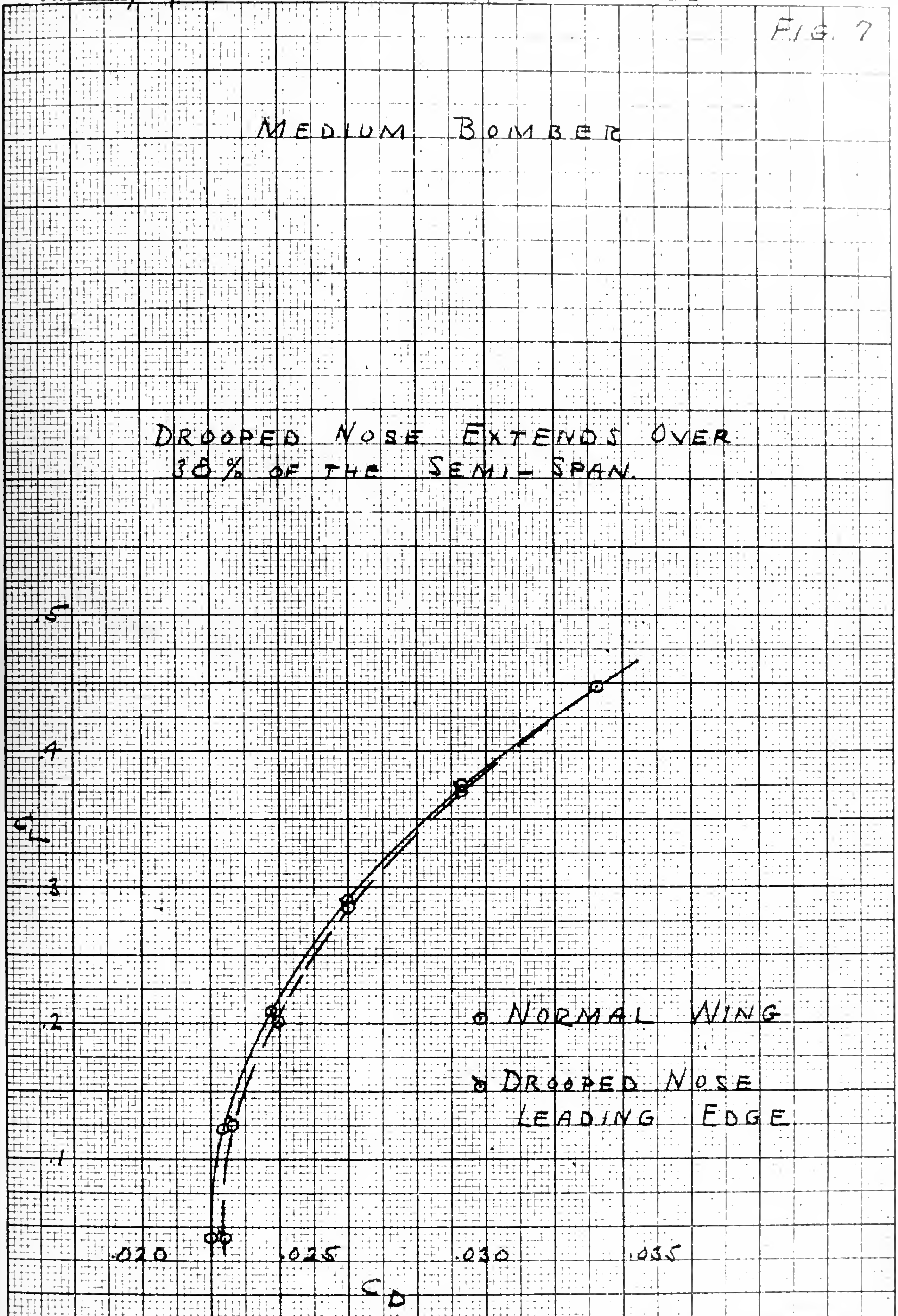
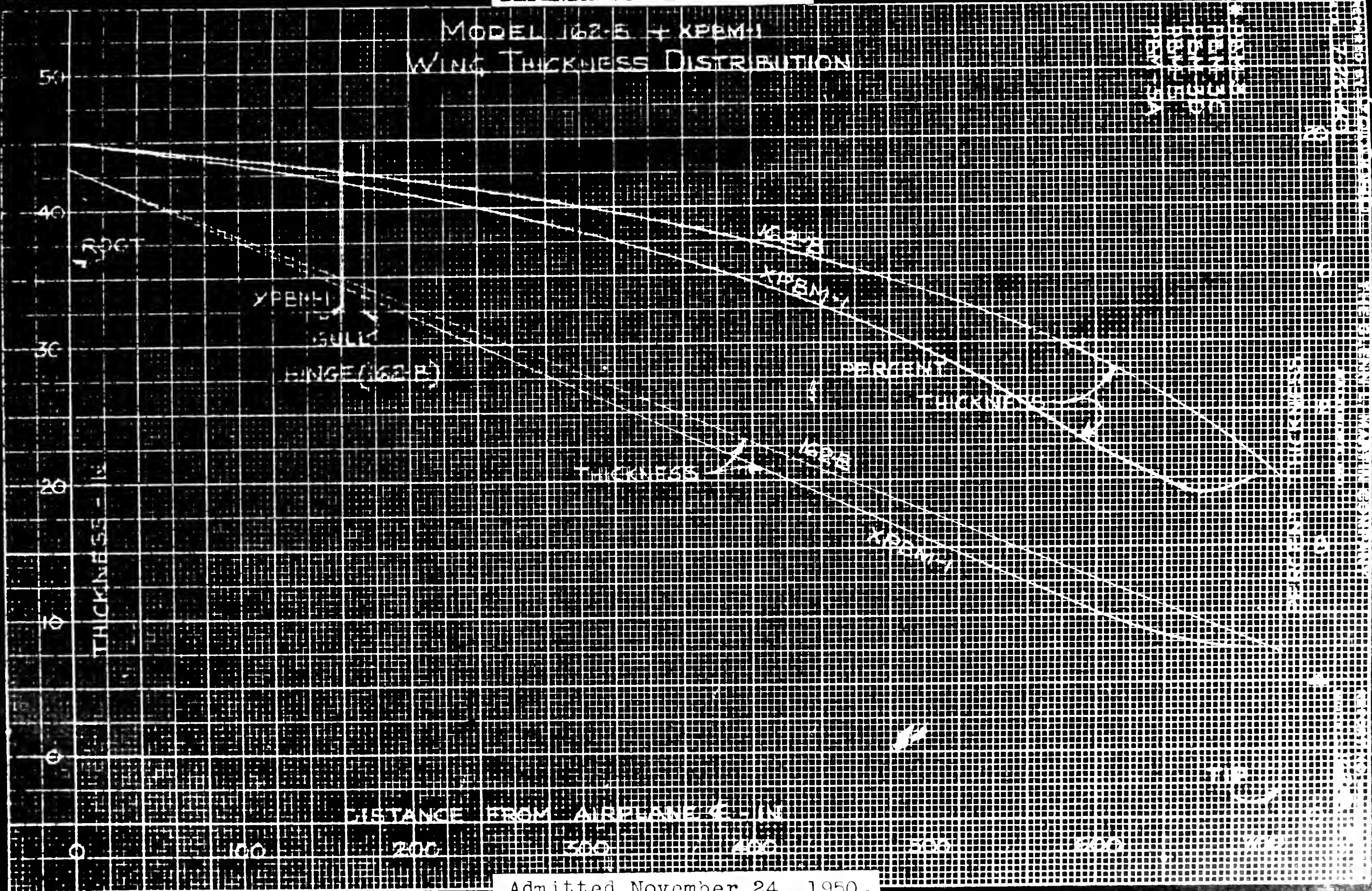


Admitted November 24, 1950.

GLM FORM 738

DEFENDANTS' EXHIBIT NN



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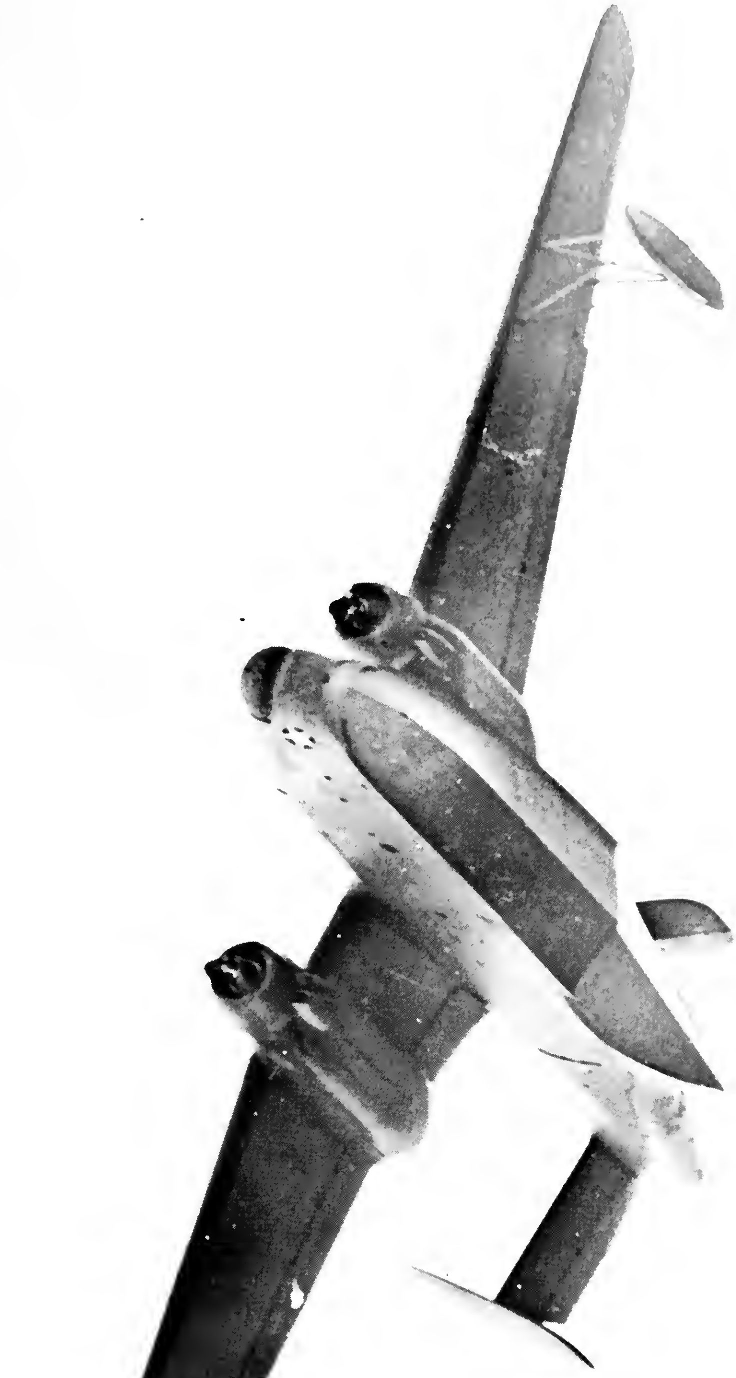

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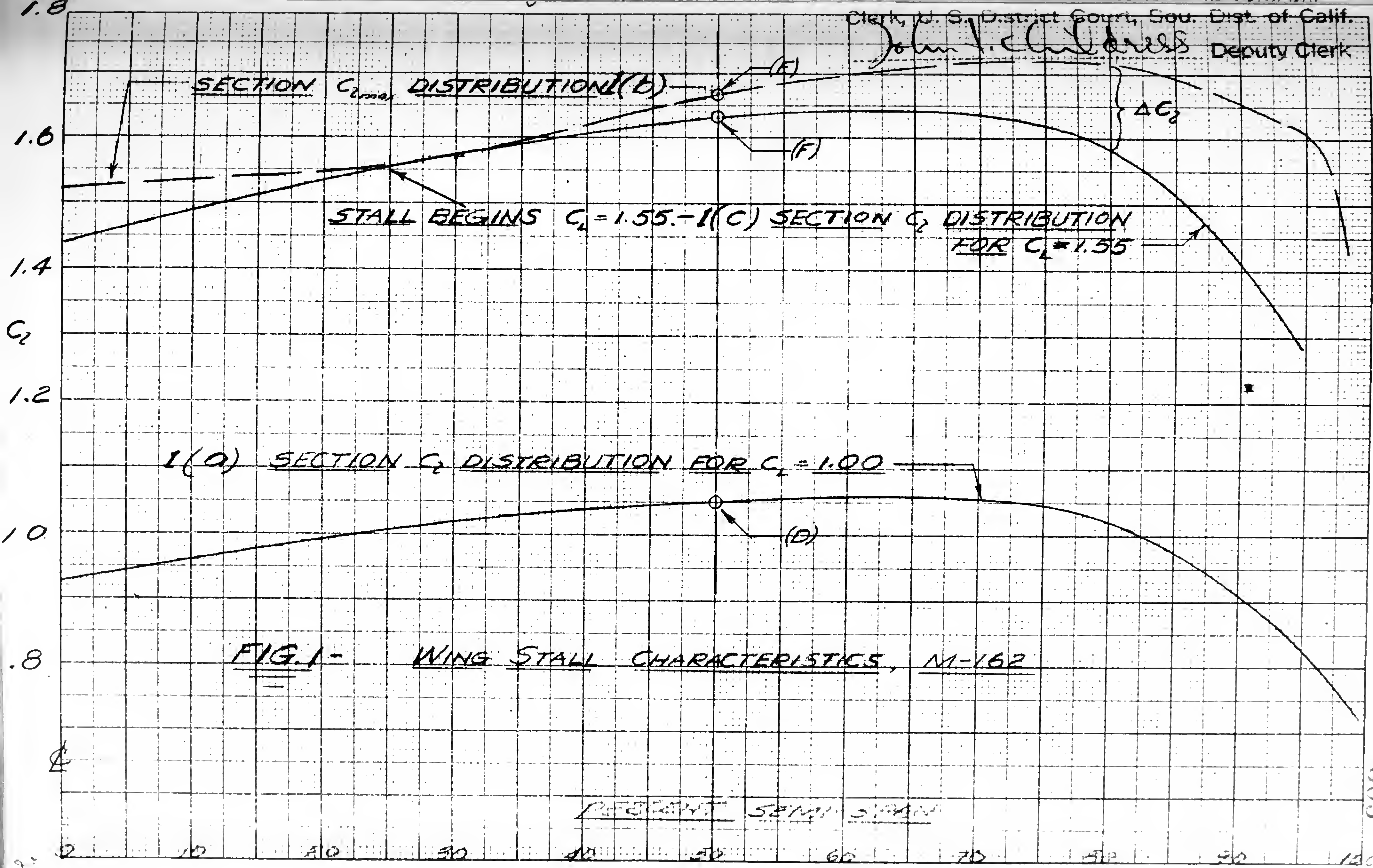

FIG. 1 – WING STALL CHARACTERISTICS, M-162

W.G. STREET

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE NO. 713

A COMPARISON OF SEVERAL TAPERED WINGS DESIGNED TO AVOID TIP STALLING

By Raymond F. Anderson

SUMMARY

Optimum proportions of tapered wings were investigat- by a method that involved a comparison of wings de- ned to be aerodynamically equal. The conditions of odynamic equality were equality in stalling speed, in uced drag at a low speed, and in the total drag at ising speed. After the wings were adjusted to aerody- ic equivalence, the weights of the wings were calcu- ed as a convenient method of indicating the optimum g. The aerodynamic characteristics were calculated m wing theory and test data for the airfoil sections. ious combinations of washout, camber increase in the foil sections from the center to the tips, and sharp ding edges at the center were used to bring about the ired equivalence of maximum lift and center-stalling racteristics.

In the calculation of the weights of the wings, a ple type of spar structure was assumed that permitted integration across the span to determine the web and flange weights. The covering and the remaining weight e taken in proportion to the wing area. The total ghts showed the wings with camber and washout to have the est weights and indicated the minimum for wings with a er ratio between 1/2 and 1/3.

INTRODUCTION

Many investigations have been made of the aerodynamic d the structural aspects of tapered wings with a view to ding the best taper ratio. Investigations of taper tio are reported in references 1 and 2. A general dis- ssion of tapered wings is given in reference 3. Although

EXHIBIT 16

ding and weight were considered in references 1 and 2, the effect of taper ratio on the maximum lift and the manner of stalling of wings was not considered. The effect of taper ratio on the maximum lift is considerable. The tip stall that usually results from the use of tapered wings, moreover, evidences itself as instability in roll at angles of attack less than that corresponding to the maximum lift coefficient. This condition is generally recognized as undesirable from the point of view of handling characteristics in low-speed flight.

It is accordingly considered herein that wings should be designed to avoid tip stalling. With this point of view, wings of different taper ratio were designed to be aerodynamically equal; that is, equal in stalling speed, in induced drag at a low speed, and in total drag at cruising speed. The weights were then calculated to indicate the "optimum" wing (the wing of lowest weight).

In the calculation of the maximum lift, the areas were so obtained that they approximate the values which would be required by wings with full-span flaps. The effect of partial-span flaps was not considered.

Wings with taper ratios of 1/2, 1/3, and 1/4 were considered for a large airplane. In the determination of the maximum lift coefficients, a margin against the stalling of the tips was specified. For the three taper ratios the stalling of three sets of wings was considered: wings with no washout or camber increase in the airfoil sections from center to tip (referred to as the "basic" series, to be described later); wings with washout; and wings with washout and camber increase from center to tips. For each of the three sets of wings, lift-spoiling devices, such as sharp leading edges, were assumed at the center of the wings to take up the required balance of the margin against stalling of the tips. This procedure is practically equivalent to increasing the lift by the use of leading-edge slots over all of the span except for a small portion of the center. The comparative effects of washout and camber should therefore be nearly independent of whether the lift is decreased at the center or increased at the tips.

ASSUMPTION FOR THE AERODYNAMIC CALCULATIONS

The wings had straight tapers and rounded tips and were of a size suitable for a four-engine airplane of 60,000 pounds gross weight with a wing loading of approximately 30 pounds per square foot. The tip chord of the trapezoid enclosing the rounded tips was used to define the taper ratio, as in reference 4. The distribution of thickness along the span and of camber and washout, when they were used, was linear. A thickness ratio of 0.09 was taken for the airfoil sections at the tips. A basic wing, used to determine the aerodynamic values to be equaled by the other wings, had a root thickness ratio of 0.14, an area of 2,200 square feet, a taper ratio of 1/3, and a span of 138.2 feet. The method of calculating the dimensions of the other wings will be given later. The symbols used are listed in an appendix.

Prevention of Tip Stalling

For the first series of wings of varying taper ratio, the method for prevention of tip stalling was the use of sharp leading edges to reduce $c_{l_{max}}$ at the center of the wings. This series of wings was called the basic series because it included the basic wing of taper ratio 1/3 used to establish the aerodynamic values. The N.A.C.A. 230 series airfoil sections listed in table I were used.

For a second series of wings, washout was used; and, for the third series, washout was combined with an increase in camber of the airfoil sections from center to tips. The increase in camber produces an increase in the $c_{l_{max}}$ of the sections near the tips and thereby causes the stalling point to move inward. For the wings with washout, small amounts of washout were used to prevent excessive increase in the induced drag. Sharp leading edges at the center of the wings were then used to make up the balance of the margin required against stalling of the tips. The case of taper ratio 1/4 was omitted for the series with washout alone because too thin a wing would have resulted.

For all the wings, in order to insure the avoidance of tip stalling, a certain c_l margin was specified at 0.7 b/2 when $C_{L_{max}}$ was reached. (See fig. 1.) The mar-

required depended on the calculated spanwise position stalling point without sharp leading edges. This occurred where a c_l curve corresponding to the se load distribution became tangent to the $c_{l_{max}}$ as outlined in detail in reference 4. When this ng point was at or inside 0.7 b/2, the c_l margin b/2 was taken as 0.1. When it was outside 0.7 b/2, argin was increased in the ratio of the distance from enter of the wing to 0.7 b/2. The provision of this of margin when stalling started at the center gave ulated positive damping in roll at the stall that prevent sudden dropping of a wing.

Conditions of Aerodynamic Equality

For the first of the conditions of aerodynamic equal- equal stalling speeds, plain airfoil sections were ed when $C_{L_{max}}$ was computed because of the availabil- f the $c_{l_{max}}$ data. The Reynolds Number at stalling was made to fall within the usual range for an air- of the size assumed by basing it on the stalling with flaps, so that the wings had approximately the reas as wings with full-span flaps. That the condi- of stalling-speed equality would not be appreciably ed by considering the wings to have full-span flaps erified from figure 60 of reference 5, which gives $c_{l_{max}}$ increments produced by flaps. (The range of verage thickness of the wings was small.)

As the stalling speed V_S is equal to $\sqrt{\dfrac{2W_g}{\rho\ S\ C_{L_{max}}}}$ W_g was fixed, the stalling-speed condition required the product $SC_{L_{max}}$ for each wing be equal to the ct for the basic wing (taper ratio 1/3).

The second condition was that the induced drags should ual at a speed corresponding to a C_L of 1.0 for the wing (low-speed condition). The induced drag rather the total drag was used because the induced drag was y all of the drag and was relatively easy to calcu- The induced drag, with the effect of twist ϵ in- d, may be found from

$$D_i = \frac{W_g^2}{q \pi b^2 u} + W_g \, \epsilon \, a_o \, v + q \, S \, (\epsilon \, a_o)^2 \, w \qquad (1)$$

o the spans required to make the induced drags equal may b expressed

$$\frac{b}{b_b} = \sqrt{\frac{u_b \, D_{i_b}}{u \, [D_{i_b} - W_g \, \epsilon \, a_o \, v - q \, S \, (\epsilon \, a_o)^2 \, w]}} \qquad (2)$$

w ere the subscript b refers to the basic wing, and

$$D_{i_b} = \frac{W_g^2}{q \, \pi \, b_b^2 \, u_b} \qquad (3)$$

Euation (3) is equation (1) with the last two terms omitt d because the basic wing has no twist. These equations w re derived from the formula for C_{D_i} given in reference

The third condition, equal cruising speeds, was satisf ed by making the drags equal at cruising speed, as the p wer was assumed constant. Cruising speed corresponded to a C_L of 0.5 for the basic wing.

METHOD OF CALCULATION

Proportions and Aerodynamic Characteristics

The method used for calculating $C_{L_{max}}$, C_{D_0}, and the o her aerodynamic characteristics of the wings has been f und to give results that agree well with test results (eferences 4 and 5).

The method of calculating the maximum lift coefficient r the basic wing is illustrated in figure 1. For this w ng, $c_l = c_{l_a}$ because there is no washout and therefore $c_b = 0$. Stalling was calculated to occur without any shar l ading edge at 0.7 b/2; that is, c_{l_a} would reach $c_{l_{max}}$ f rst at the 0.7 point. (See reference 4 for a detailed

explanation.) A value of c_{l_a} of 0.1 less than the $c_{l_{max}}$ at $y = 0.7\ b/2$ (c_{l_a}') was then the lift coefficient corresponding to $C_{L_{max}}$. Numerically, $C_{L_{max}} = c_{l_a}'/c_{l_{a1}}$, where $c_{l_{a1}}$ was taken at $y = 0.7\ b/2$. The values of $c_{l_{max}}$ at the center of the wing were then considered to be reduced by a sharp leading edge to the values of c_{l_a}, as shown, so that stalling would begin at the center of the wing. The values of $c_{l_{max}}$ used for calculating $C_{L_{max}}$ for this wing were taken from reference 5.

The value of the induced drag at the low-speed condition for the basic wing, D_{i_b}, to be used in finding the spans of the other wings was calculated from equation (3).

The drag of the basic wing at cruising speed was calculated in terms of q in the form

$$\frac{D}{q} = \frac{D_o}{q} + \frac{D_i}{q} \qquad (4)$$

The value of D_o/q was calculated for a C_L of 0.3 and for the cruising-speed Reynolds Number (as outlined in reference 4) by a graphical integration along the span of the section drags from

$$\frac{D_o}{q} = \int_0^{b/2} c_{d_o}\ c\ dy \qquad (5)$$

The values of c_{d_o} were taken from reference 7 for the basic wing as well as for the others. The value of D_i/q was calculated from equation (3) for a value of q corresponding to the cruising speed.

With the values for the basic wing established, equal values for the other wings were found by successive approximations. For the other two wings of the basic series, a root thickness and an area were assumed that, it was hoped, would produce the desired characteristics. An approximate

span was then found from equation (2) so that c and c_{l_a} could be found. For these values, $C_{L_{max}}$ was then calculated in the same manner as for the basic wing.

For the wings with washout and with washout and camber increase, airfoil sections and washout were assumed. The value of $C_{L_{max}}$ was then calculated as for the basic wing, except that c_{l_b} due to washout was combined with c_{l_a} to obtain c_l, as shown in figure 2.

From the values of $C_{L_{max}}$ for the wings, a more accurate value of S was found for each wing to obtain a product of S and $C_{L_{max}}$ equal to the value for the basic wing. The approximate span was used to calculate the aspect ratio so that the induced-drag factors u, v, and w could be found from reference 4. A more accurate value of the span to obtain the required induced drag at low speed could then be found from equation (2). A value of a_0 of 0.1 per degree was used. From S and b, more accurate values of c could be found so that D/q could be computed.

The value of D/q at cruising speed for each wing was next found from equation (4), where the value of D_0/q was calculated from equation (5) for a C_L corresponding to the cruising speed and the wing area. The value of D_i/q was then found from equation (1) for a value of q corresponding to the cruising speed. If the values of D/q calculated in this manner were not close to the value for the basic wing, new values of root thickness ratio were assumed and the calculations were repeated.

Successive approximations were repeated in this manner until the required values of $SC_{L_{max}}$, b, and D/q were obtained. Two or three approximations were usually required. The resulting dimensions and the values of D/q are given in table I. The amounts of washout required were a compromise between a high $C_{L_{max}}$ and a low induced drag. In order to investigate the effect of greater washout, calculations were made for a wing with camber increase and washout with a taper ratio of 1/3, and with $\epsilon = -4^\circ$, but the results were not included in the table because the weight was excessively increased. It should be noted that

the washout is "aerodynamic"; that is, it is measured, not from the chord, but from the zero-lift directions of the root and the tip sections.

Weight of the Wings

The load factors for calculating the weights of the wings were computed as specified in reference 8. A high speed of 240 miles per hour was used with a gust of 30 feet per second, as given for condition I in reference 8. The lift-curve slope was computed from figure 2 of reference 4. The values of the limit-load factors n, computed in this manner, are listed in table I.

The C_N to be used for calculating the load on the wings was then found from

$$C_N = \frac{n(W_g - W)}{qS} \tag{6}$$

where W_g is the gross weight; W, the assumed wing weight; and q corresponds to a speed of 240 miles per hour. The load distribution per unit length along the span, l, was then found from $l = q\, c_l\, c$ where c_l was found as in reference 4 from

$$c_l = C_N\, c_{l_{a1}} + c_{l_b} \tag{7}$$

For the wings without twist, c_{l_b} is zero.

The values of $c_{l_{a1}}$ and c_{l_b} were calculated from the load-distribution data given in reference 4 so that the variation of the load distribution with taper was taken into account. From the distribution of load across the span, the distribution of the shear and the moment could be easily found.

The shears and the moments were assumed to be carried by a single spar with a simple type of structure as shown in figure 3, so that the weights of the material could be

nt by an integration across the span. The torsion load
eliminated by assuming the spar to be located at the
t center of each section may be considered to be car-
d by the skin.

The relieving loads caused by the engines and the fu-
age were taken into account so that the total wing
hts were calculated in the form

$$W = W_W - \Delta W_W + W_F - \Delta W_F + W_C \qquad (8)$$

The weights thus calculated may not agree with the
hts of actual airplane wings because of the simple
of structure assumed and the improbability that all
material will develop the stress assumed. The effects
he assumptions should, however, be similar on all the
s so that the correct relative weights should be ob-
able.

The load distributions across the semispan of the
s, computed in the manner previously given, had the
represented in figure 3. From the load, or $c_l c$,
es, the shears and the moments at any point y along
semispan were found from

$$F_S = q \int_y^{b/2} c_l c \, dy \qquad (9)$$

$$M = \int_y^{b/2} F_S \, dy \qquad (10)$$

shear bracing was assumed to have an angle of 45°, as
n in figure 3. For a unit length along the span dy
esponding to a unit length of bracing dL, the weight
he web will be

$$dW_W = p \frac{f}{s} dL = p \frac{F_S}{0.707 s} \frac{dy}{0.707} = \frac{2p F_S}{s} dy \qquad (11)$$

e

p is the specific weight (assumed to be an aluminum alloy weighing 0.1 pound per cubic inch).

s, allowable stress.

f, force in a diagonal.

h factor of safety of 1.5, the web weight for both es of the wing is then

$$W_W = 4 \times 1.5 \frac{p}{s} \int_0^{b/2} F_S \, dy \tag{12}$$

cnservative stress of 20,000 pounds per square inch was smed in calculating W_W.

In the calculation of the weight of the flanges, the nt at any point along the span was considered to be ried by tension and compression in the flanges. If F he force in a flange (fig. 3) and if the effective ikness of the beam t' is taken as 0.9 the wing thick-s, then the weight of a unit length of one flange will

$$dW_F = p \frac{F}{s} dy = p \frac{M}{t's} dy \tag{13}$$

weight of upper and lower flanges for both halves of wing, with a factor of safety of 1.5, is then

$$W_F = 4 \times 1.5 \frac{p}{s} \int_0^{b/2} \frac{M}{t'} dy \tag{14}$$

rn equations (12) and (14), the web and the flange ghts were found by graphical integration of curves of S and M/t' along the semispan. Values of s of 20,000 onds per square inch for compression and 30,000 pounds e square inch for tension were used to calculate the lnge weights.

In the calculation of the weight decrements due to the eieving loads, the concentrated loads shown in figure 3 ere considered, and the useful loads were omitted to be con-crvative. The shear was assumed to be taken off at the

lage wall so that half the weight of the body $W_B/2$ at a distance y_B. The weight of the body consists he complete weight of the fuselage and the tail, less useful load. The nacelles and the cowling were includ- h the power-plant weights, W_{P_1} and W_{P_2}, and the ing-gear weight was included in W_{P_1}. The correct rel- e weights of the relieving loads were established by a nt analysis.

The relieving effect of each load on the web weight roportional to the load times its distance from the er. Then, from equation (11), the web-weight decrement both halves of the wing, with a factor of safety of 1.5 a limit-load factor n, may be written

$$\Delta W_W = \frac{4 \times 1.5\ pn}{s} \left(\frac{W_B}{2} y_B + W_{P_1} y_1 + W_{P_2} y_2\right) \qquad (15)$$

e same value of s was used as in the web-weight calcu- on.

The relieving effect of each load on the flange weight roportional to the moment times the distance of the a from the center. Then if $t_s{}'$ is 0.9 the root thick- s, the weight decrement due to the relieving loads for flanges and both halves of the wing will be, from equa- (13),

$$\Delta W_F = \frac{4 \times 1.5\ pn}{t_s{}'\ s} \left(\frac{W_B}{2} y_B{}^2 + W_{P_1} y_1{}^2 + W_{P_2} y_2{}^2\right) \qquad (16)$$

e same values of s were used as for the flange-weight ulation.

The final weight item W_C, which included the cover- g and all of the structural weight other than that of beam, was taken as a constant proportion of the wing . The net weights of the various structural parts of wing and the total weights are listed in table I. As wing weight was found, it was compared with the as- d weight used in equation (6) and the calculations repeated until the value of the weight assumed did not ct the final weight.

RESULTS AND DISCUSSION

From the dimensions and the characteristics of the wings listed in table I, the effect of changes of the taper and of the method to prevent tip stalling may be noted. The effect of a change of the taper on $C_{L_{max}}$ and on the resulting area may be explained as follows. As the taper is increased, c_l increases from the center to the tip of the wing. In addition, the Reynolds Number decreases toward the tips so that, for the usual airfoil sections, $c_{l_{max}}$ decreases. The value of $C_{L_{max}}$ is thereby reduced and stalling tends to start nearer the tips. A greater amount of the means to prevent stalling of the tips must therefore be used to obtain the desired c_l margin, as the taper is increased. The amount required may be measured in terms of the difference, at the center of the wing, between $c_{l_{max}}$ and the c_l corresponding to $C_{L_{max}}$ (shown by Δc_l in fig. 1). Thus, Δc_l increases with taper, as listed in table I. Because of the foregoing effects, the areas also tend to increase with the taper, as shown in table I.

The change in span required to obtain the desired induced drag for the low-speed condition depends only on the value of the induced-drag factor u for wings without twist. As the value of u, which is a measure of the change of induced drag with taper for wings without twist, changes only slightly with the taper, the span varies only slightly, as shown in table I. The wings with washout, however, require a greater change in span owing to the twist, as may be seen from equation (2) and as given in the table.

The increase in area with increase in taper previously mentioned requires a reduction in thickness to obtain the required low value of the profile drag at the cruising condition. The exact value of profile drag required also depends on the induced drag at cruising speed, as the total drag must have a fixed value. This induced drag tends to be adversely affected by an increase in taper or in washout. The combined effect of washout and taper is appreciable for the wings with washout and camber increase, as shown by the values of D_i/q in the table. The foregoing effects cause the required thickness to decrease with the taper.

When the thickness was changed to make another approx- ion in the calculation of the characteristics of the s, $C_{L_{max}}$ was affected as well as the drag. Whether change increased or decreased $C_{L_{max}}$ depended on the kness ratio near 0.7 b/2 and on the corresponding x. The effect may be predicted for any particular from figure 55 of reference 6, which shows the varia- of $c_{l_{max}}$ with thickness ratio. A decrease in root kness ratio usually increased $C_{L_{max}}$.

For the wings with camber increase, the increase in ber toward the tips increased $c_{l_{max}}$ and produced er $C_{L_{max}}$ values and lower areas. As some sharp lead- edge was used for all the wings to obtain the desired margin, the wings should be comparable in their avoid- of tip stalling.

For the wings with washout and camber increase, the red margin could have been obtained by more washout the induced drag would have been too greatly increased. l amounts of washout were used, as listed, and the cam- was increased from 3 to 4 percent of the chord as the r ratio changed from 1/2 to 1/3. No further increase amber for the wing of taper ratio 1/4 was used because would have produced no further increase in $c_{l_{max}}$.

With reference to the weights of the wings, it may be d that the lowest weights were obtained for the wings camber increase and washout. The lowest weight is in- ated for a taper ratio between 1/2 and 1/3, as may be from figure 4. In order to determine whether the low- weight had been approached, the case of taper ratio with washout and camber increase was investigated with ce as much washout, or 4°. The increase in washout re- red a reduction in thickness to obtain the desired drag cruising speed and an increase in span to maintain the ired induced drag at low speed. The result was a con- erable increase in weight.

If this analysis were applied to wings of other size, ax and D_0 would be affected by the change in Reynolds ber, but it is believed that considerable variation in e would be possible without altering the conclusion as the best taper ratio. The number of engines is also of

ight importance because the effect of their relieving
ad on the wing weight is small. It is also believed
at, for the thickness ratios in common use, the selec-
on of a different thickness ratio for the basic wing
uld not appreciably alter the conclusions.

As an aid in similar calculations and to show the ef-
ect of washout on C_{D_i}, the change in C_{D_i} due to wash-
t has been plotted in figures 5 to 7. The increase in
C_{D_i} may be considered to consist of two parts, which may
found by dividing the last two terms of equation (1) by
. The $w(\epsilon a_0)^2$ term is the increase in C_{D_i} for C_L =
and varies mainly with ϵ^2, as w does not vary much
the usual range of taper ratios. (See fig. 6 of ref-
ence .) The term $v \epsilon a_0 C_L$ contributes a positive or
negative increment depending on the sign of v except
at, for the elliptical wing, $v = 0$ and ΔC_{D_i} does not
ry with C_L. For the tapered wings, however, ΔC_{D_i} in-
eases with C_L for taper ratios less than about 1/2, as
ay be seen from figures 5 to 7.

For taper ratios approaching 1, ΔC_{D_i} becomes nega-
ive for high values of C_L as shown by figure 7, which
eans that an elliptical span loading is approached owing
o the washout. Values of ΔC_{D_i} for other aspect ratios
nd taper ratios, for either washin or washout, may be cal-
ulated from reference 4.

The values of ΔC_{D_i} given are for wings with linear
wist distribution along the span. Wings are commonly
onstructed using straight-line elements between corre-
sponding points of the root and the tip sections. For such
construction, the twist distribution is nonlinear and,
for a given washout at the tip, ΔC_{D_i} is less than for a
inear twist distribution. As an illustration of the order
of magnitude of the difference that the type of twist dis-
ribution may produce, values of ΔC_{D_i} are given in fig-
ure 8 for wings with trapezoidal tips and with the two
types of twist distribution. As may be seen, the differ-
ences are small. With reference to the effect of the type
of twist distribution on the lift distribution, and hence
on the margin against stalling of the tips, it may be said
that the amount of washout required is substantially the

for the two types of twist distribution for taper ratios between 1/3 and 1.0.

From the present paper and from the data given in reference 4, similar calculations can be made for wings of any size and for any aerodynamic conditions. Analyses should probably be made for wings with partial-span flaps and other high-lift devices.

CONCLUSIONS

For wings within the range of thickness ratios commonly used, designed to be aerodynamically equal, and with tip stalling avoided by the methods considered, the results of this analysis indicate that:

1. The optimum wings (the wings of the lowest weight) are obtained when tip stalling is prevented by the use of moderate washout combined with an increase in camber of the airfoil sections from the center to the tip.

2. The optimum wings have a taper ratio between 1/2 and 1/3.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., May 3, 1939.

APPENDIX

Symbols

S, wing area.

b, span.

b_b, span of basic wing.

A, aspect ratio, b^2/S.

c, chord at any section along the span.

ϵ, aerodynamic twist, in degrees, from root to tip, measured between the zero-lift directions of the center and the tip sections, negative for washout.

y, distance along the span measured from the center.

y_2, see figure 3.

a_0, section lift-curve slope, per degree.

c_l, section lift coefficient; $c_l = c_{l_a} + c_{l_b}$.

c_{l_b}, part of lift coefficient due to aerodynamic twist (computed for $C_L = 0$); $c_{l_b} = \frac{\epsilon a_0 S}{cb} L_b$.

c_{l_a}, part of lift coefficient due to angle of attack at any C_L; $c_{l_a} = C_L c_{l_{a1}}$.

$c_{l_{a1}}$, part of lift coefficient due to angle of attack for $C_L = 1.0$; $c_{l_{a1}} = \frac{S}{cb} L_a$.

L_a, L_b, additional and basic load distribution parameters (Values of L_a and L_b were taken from reference 4 to obtain the load distributions.)

$c_{l_{max}}$, airfoil section maximum lift coefficient.

c_{d_0}, airfoil section profile-drag coefficient.

C_N, wing normal-force coefficient (taken equal to C_L).

C_L, wing lift coefficient.

$C_{L_{max}}$, wing maximum lift coefficient.

C_{D_o}, wing profile-drag coefficient.

C_{D_i}, wing induced-drag coefficient.

ΔC_{D_i}, increase in wing induced-drag coefficient due to aerodynamic twist.

D, total wing drag.

D_o, wing profile drag.

D_i, wing induced drag.

D_{i_b}, induced drag of the basic wing.

and w, induced-drag factors (reference 4).

n, limit-load factor.

l, load distribution per unit length along the span.

W_g, airplane gross weight.

W, wing weight.

Subscripts W, F, and C refer to web, flange, and cover weights, respectively.

Δ refers to a weight decrement due to relieving loads.

F_S, shear force at any point along the span.

M, bending moment at any point along the span.

p, specific weight (of aluminum alloy, 0.1 lb./cu. in.).

s, allowable stress.

t', effective thickness of beam at any point along span.

t_s', effective thickness of beam at center of wing.

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obtained by –	ratio	(sq.ft.)	ratio, A	(ft.)	chord c_s (ft.)	chord c_t (ft.)	airfoil section N.A.C.A.	airfoil section N.A.C.A.		(deg.)
Sharp leading edge	1/2	2,132	8.93	138.0	20.91	10.46	23015.4	23009	0.16	0
	1/3	2,200	8.68	138.2	24.20	8.07	23014	23009	.36	0
	1/4	2,350	8.16	138.5	27.29	6.82	23011	23009	.43	0
Washout and sharp leading edge	1/2	2,090	9.13	138.1	20.48	10.24	23015.5	23009	--	2
	1/3	2,194	8.77	138.8	23.89	7.96	23013.2	23009	--	1
Washout, camber increase, and sharp leading edge	1/2	2,082	9.15	138.0	20.40	10.20	23016	33009	--	1
	1/3	2,080	9.32	139.2	22.55	7.52	23014	43009	--	2
	1/4	2,149	9.16	140.2	24.62	6.16	23012	43009	--	2

([a]Sections with sharp leading edge.)

Center stall obtained by –	D_0/q	D_i/q	D/q	C_L at cruising speed	Wing loading W/S	Limit-load factor, n	Weight of flanges, $W_F-\Delta W_F$ (lb.)	Weight of web $W_W-\Delta W_W$ (lb.)	Weight of covering and bracing W_C (lb.)	Total Weight W (lb.)
Sharp leading edge	12.7	7.4	20.1	0.309	30.0	2.98	5,930	624	2,533	9,087
	12.8	7.4	20.2	.300	29.1	3.04	5,572	615	2,614	8,801
	12.7	7.4	20.1	.281	27.2	3.15	6,202	607	2,790	9,599
Washout and sharp leading edge	12.6	7.6	20.2	.316	30.6	2.93	5,465	582	2,481	8,528
	12.5	7.5	20.1	.300	29.2	3.03	5,656	592	2,606	8,854
Washout, camber increase, and sharp leading edge	12.7	7.4	20.1	.317	30.7	2.91	5,521	599	2,474	8,594
	12.4	7.8	20.2	.317	30.8	2.93	5,395	568	2,470	8,433
	12.3	7.9	20.2	.307	29.8	2.98	5,904	565	2,551	9,020

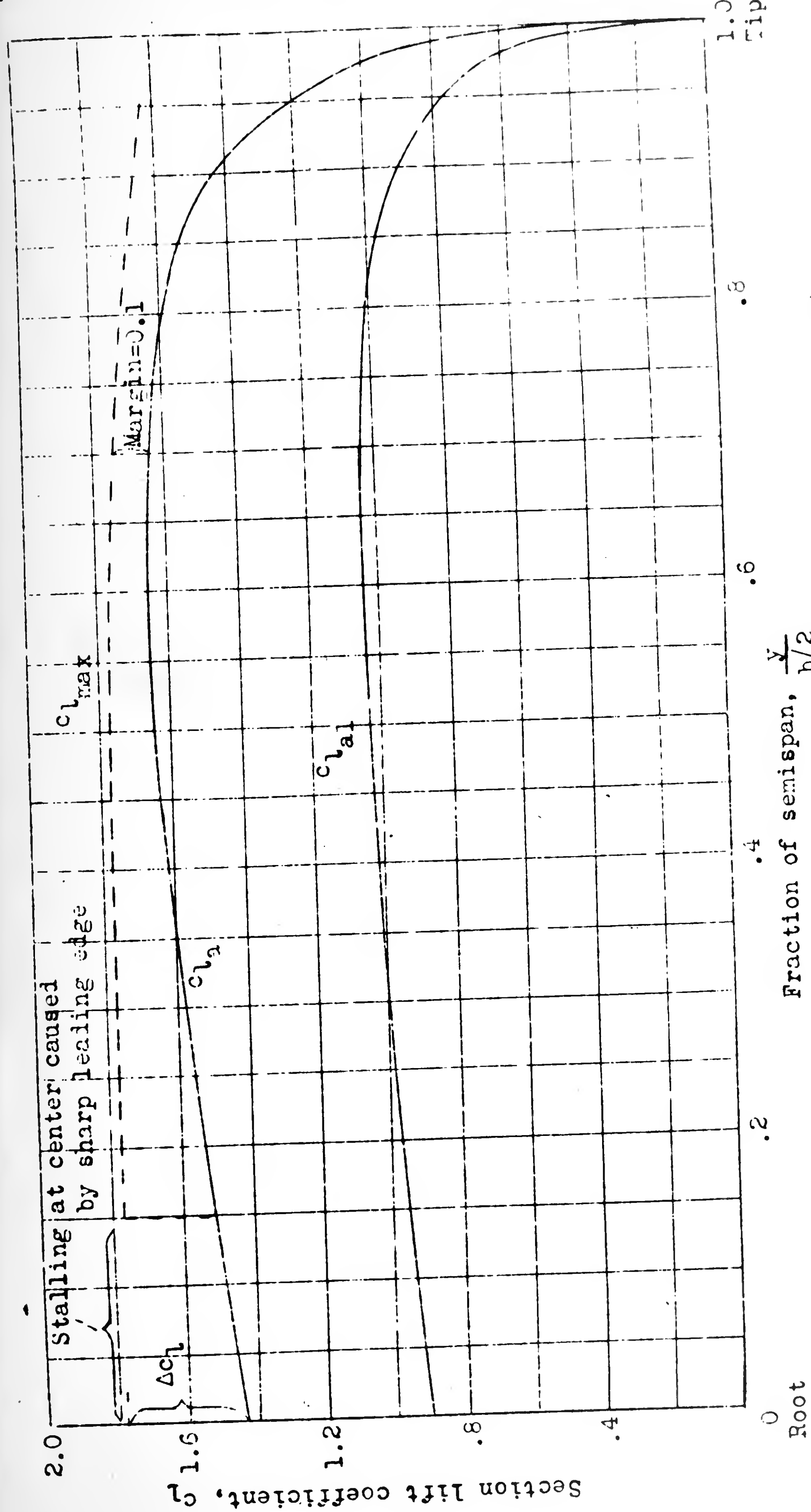

Figure 1.- Calculated stall of wing with sharp leading edge; taper ratio, 1/3 .

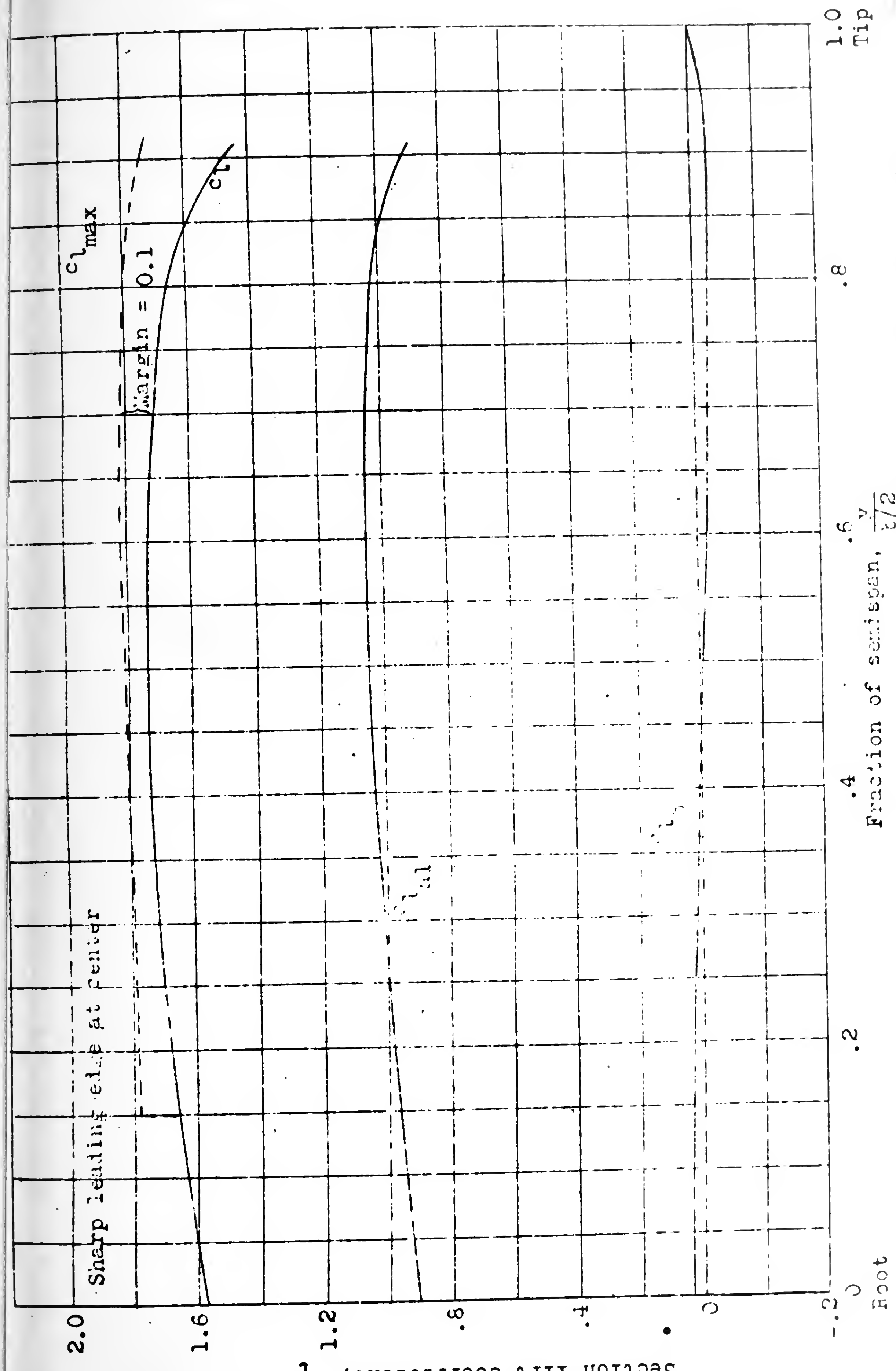

Figure 2.- Calculated stall of wing with camber increase and washout; taper ratio, 1/3.

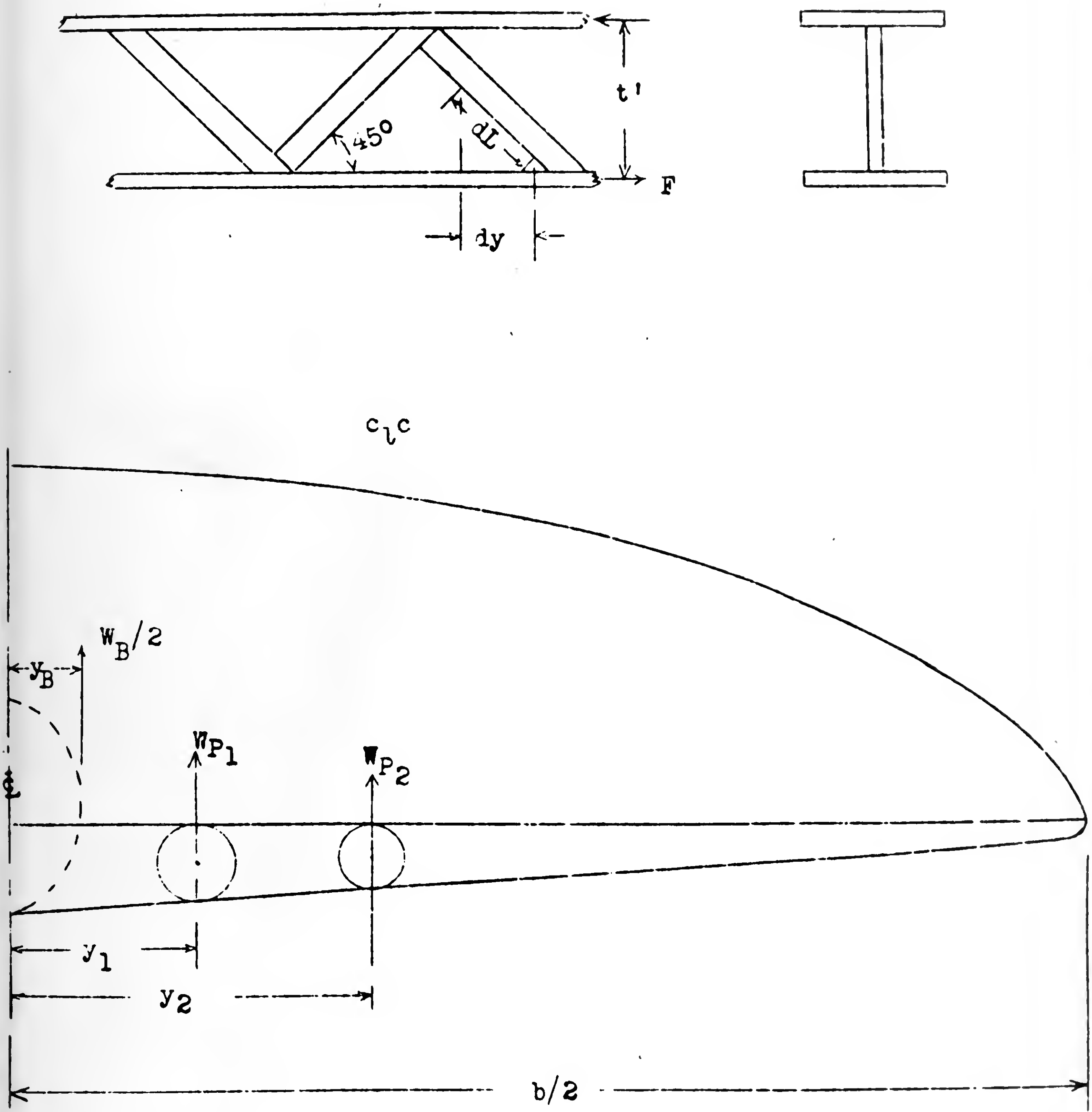

Figure 3.- Spar structure and loads on wing.

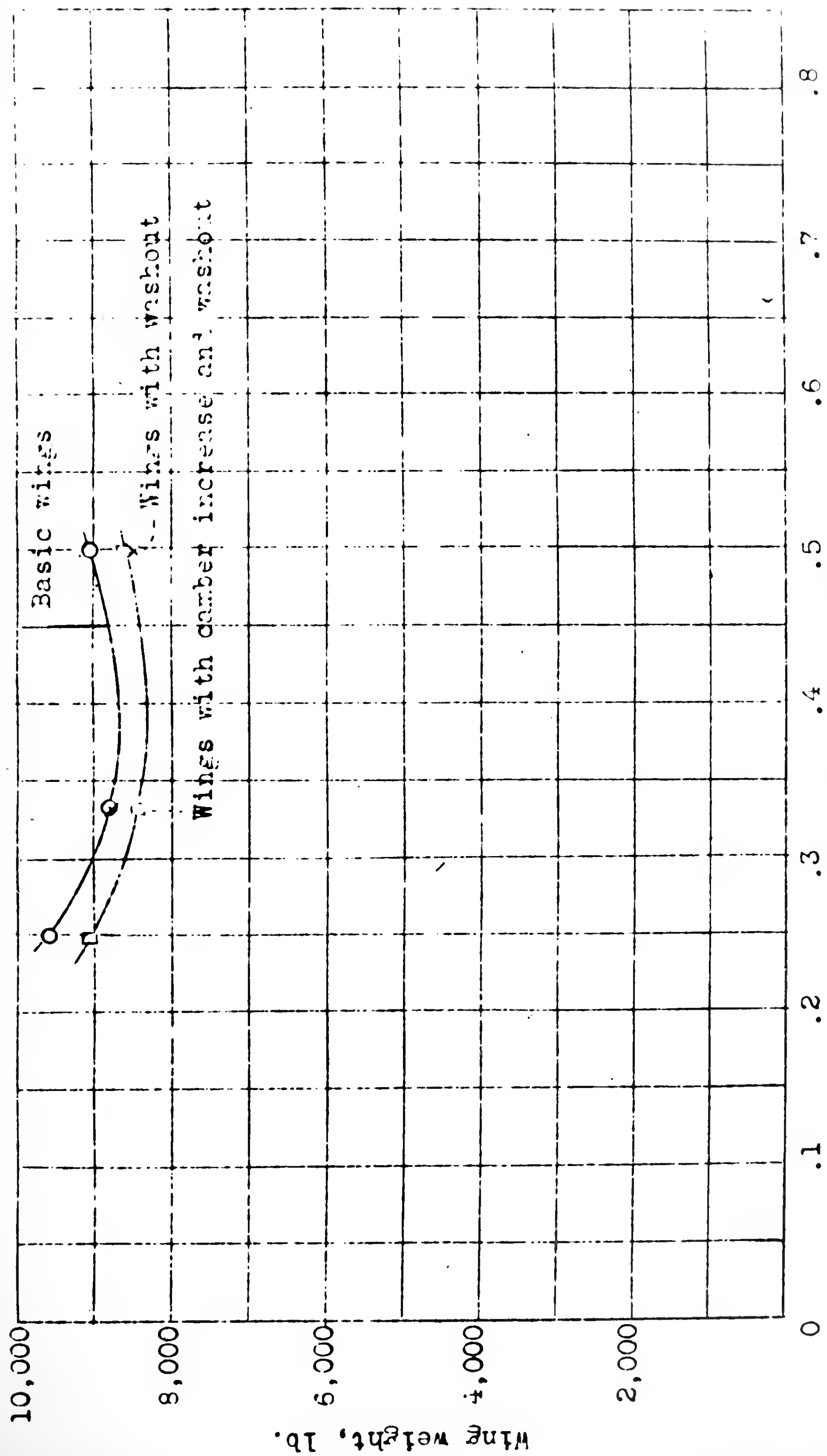

Figure 4.- Variation of wing weight with taper ratio.

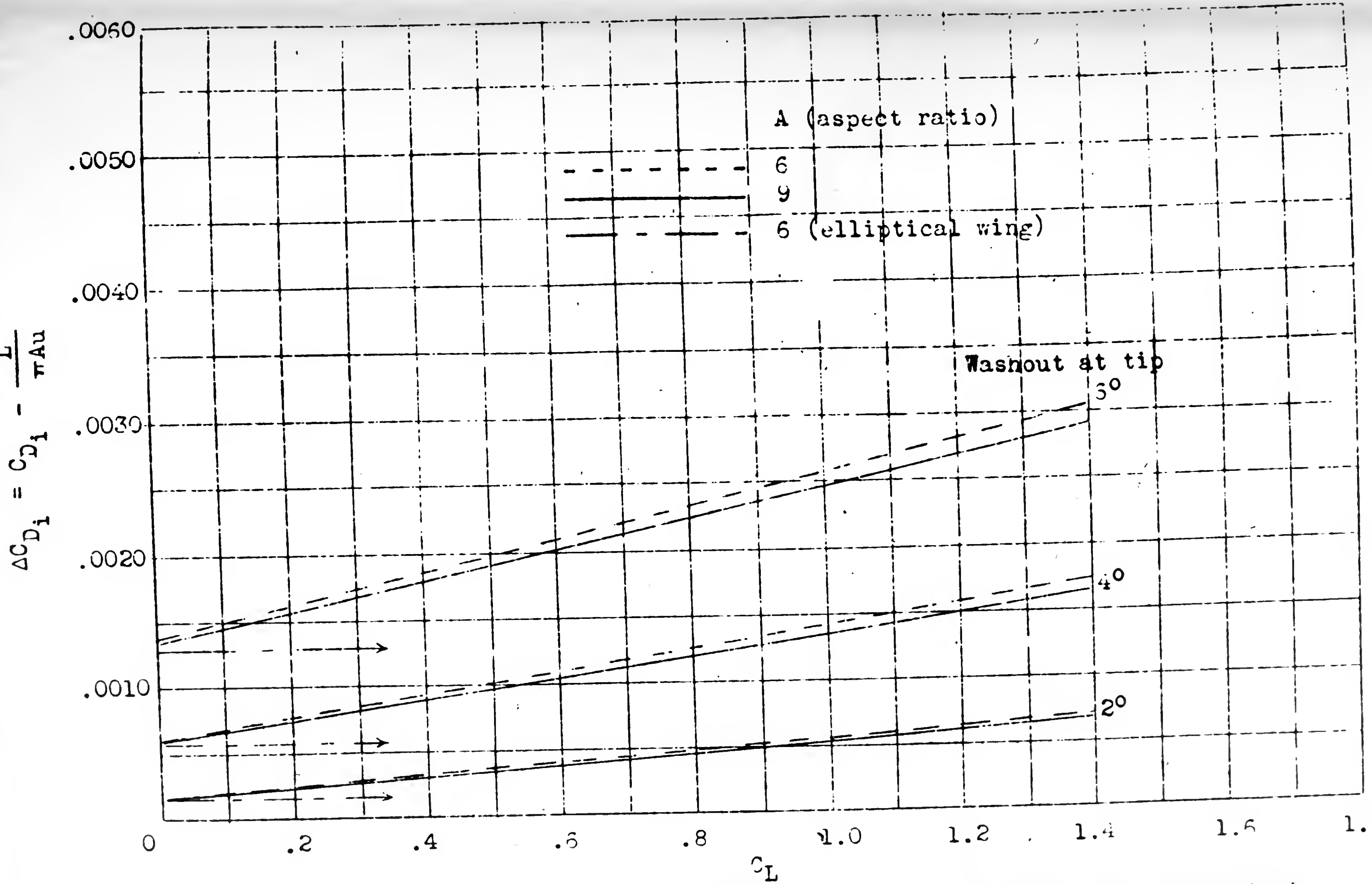

Figure 5.- Increase in induced-drag coefficient due to linear washout; rounded-tip wings; taper ratio, 1/3.

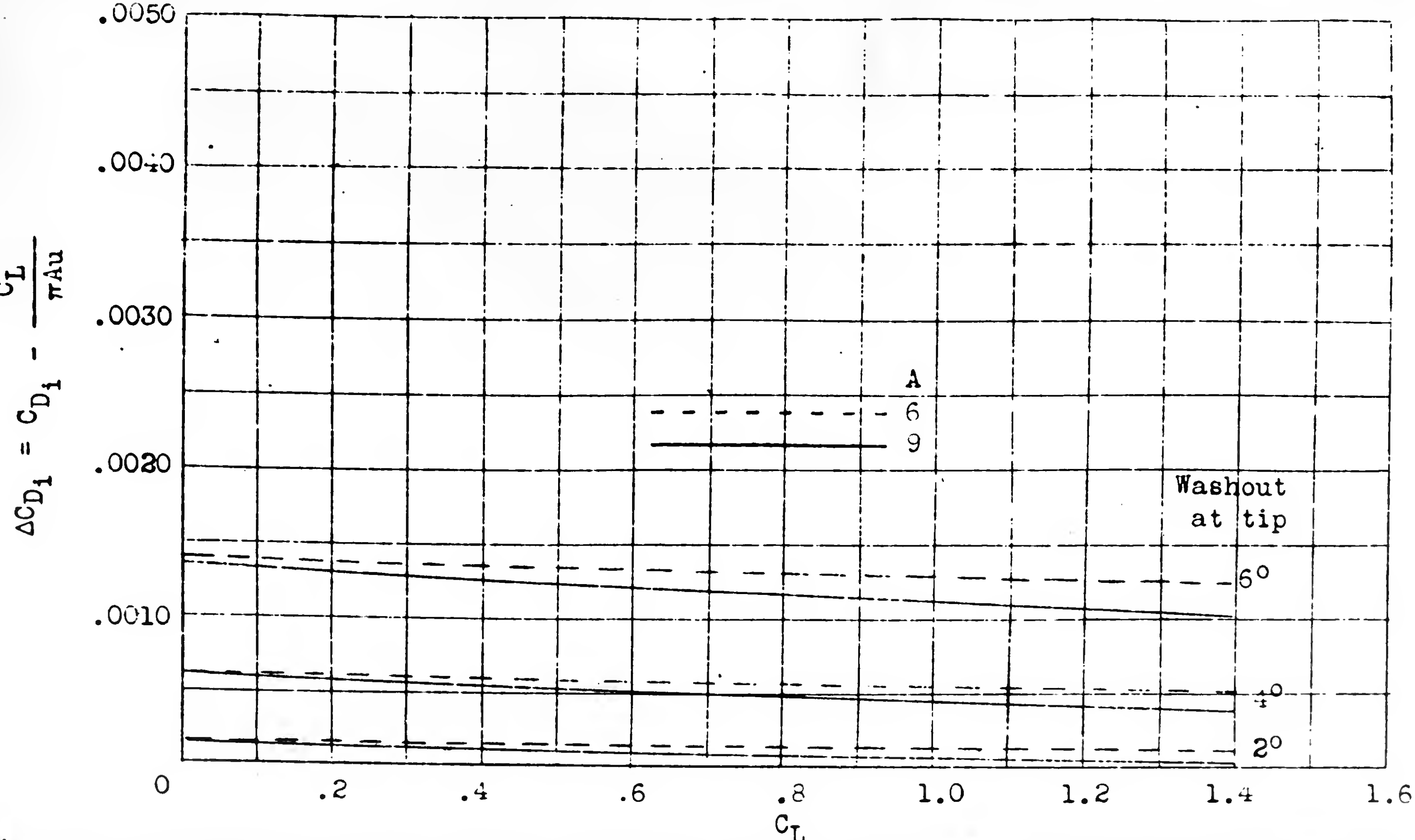

Figure 6.- Increase in induced-drag coefficient due to linear washout; rounded-tip wings; taper ratio; 1/2 .

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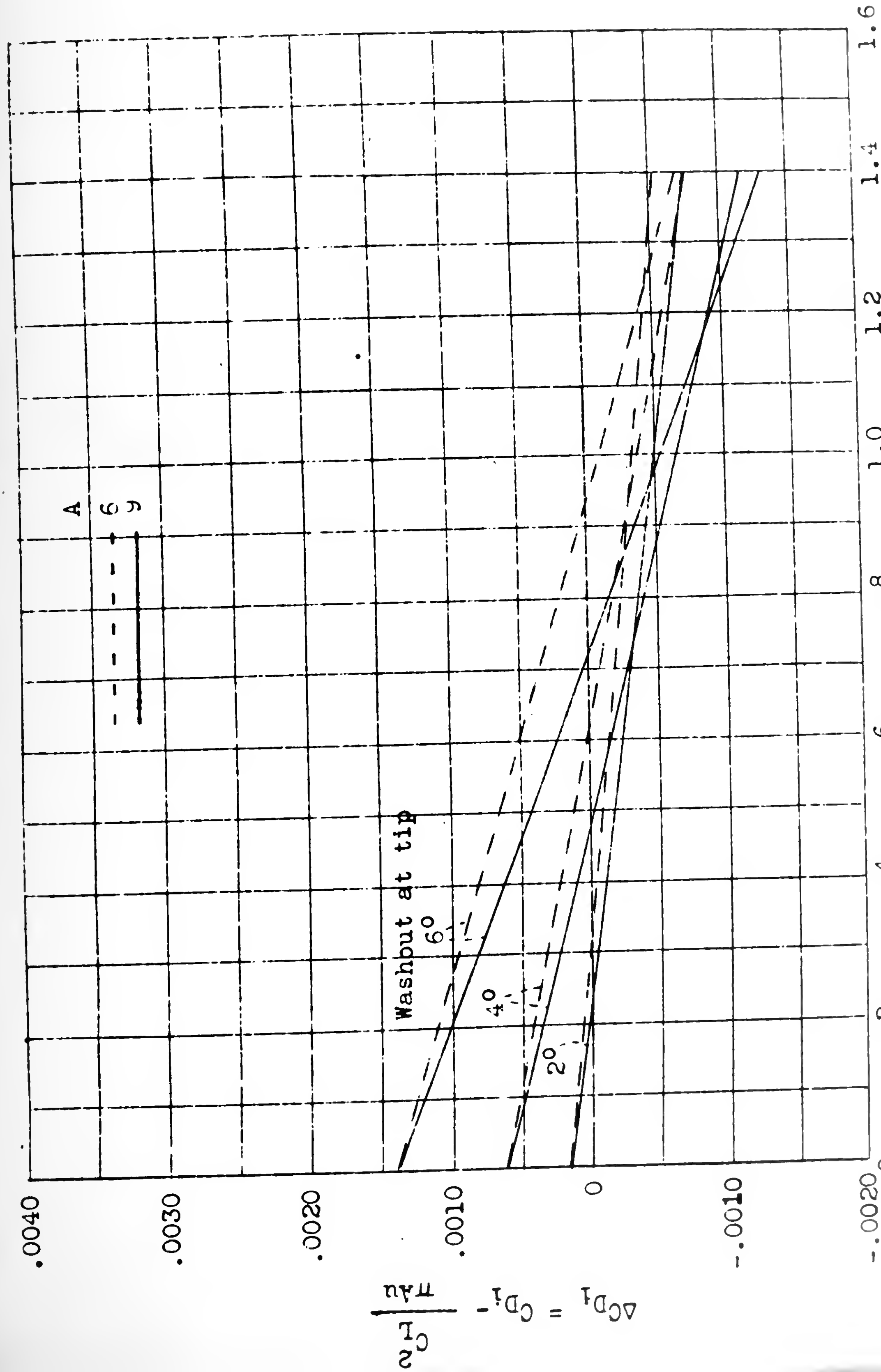

Figure 7.- Change in induced-drag coefficient due to linear washout; rounded-tip wings; taper ratio, 3/4.

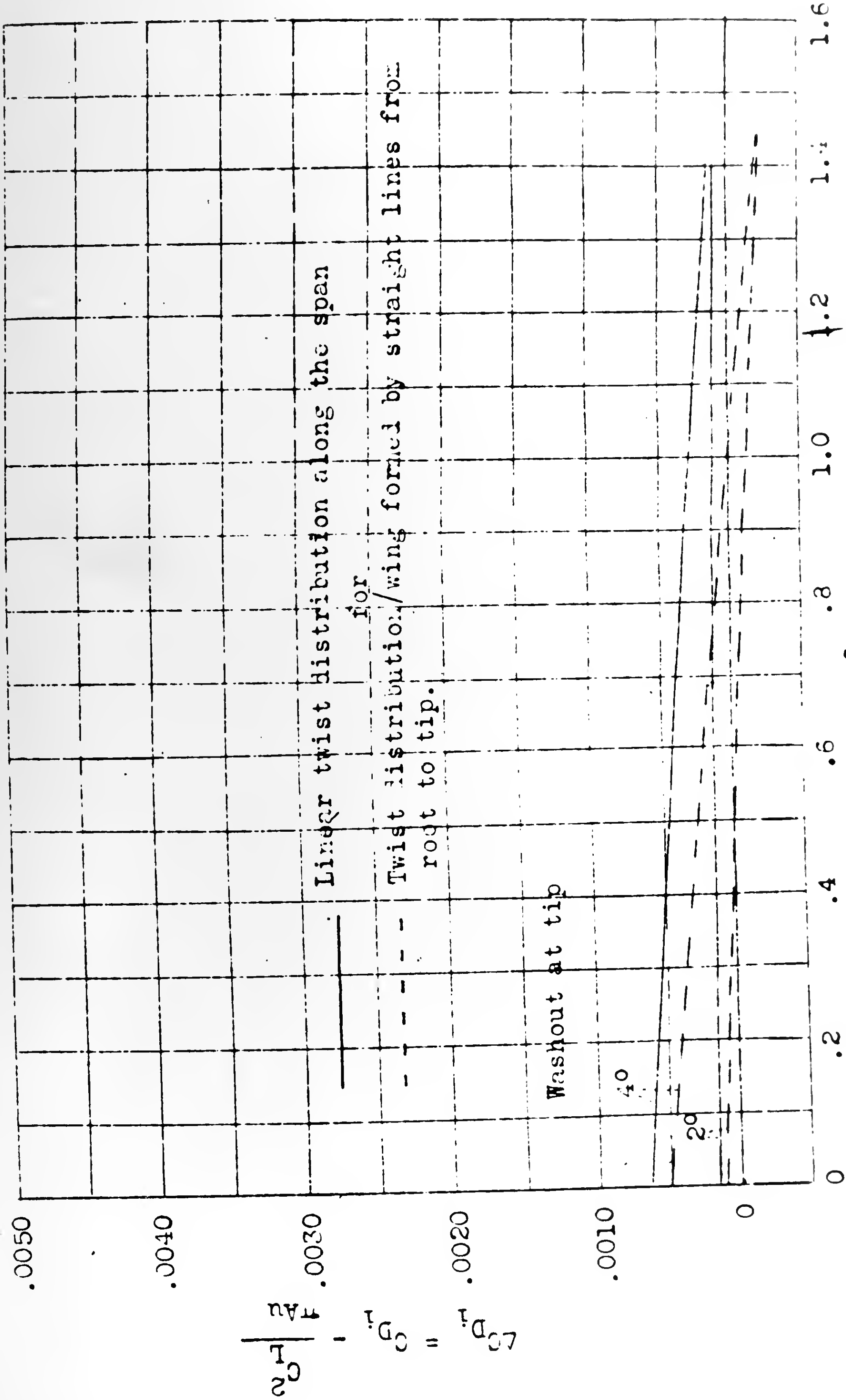

Figure 8.- Effect of type of twist distribution on ΔC_{D_i} : wings with trapezoidal tips: aspect ratio. 6 ; taper ratio, 1/2 .

Admitted November 24, 1950.

DEFENDANTS' EXHIBIT VV

District Court of the United States, Southern District of California, Central Division
Civil Action No. 10930-Y

MAURICE A. GARBELL, INC., a California Corporation, and GARBELL RESEARCH FOUNDATION, a California Corporation,
Plaintiffs,

vs.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION, a Delaware Corporation, and AMERICAN AIR LINES, INC., a Delaware Corporation,
Defendants.

STIPULATION #2

It is hereby stipulated subject to proof of error that the appended "Exhibit 18" is a reproduction of pages 267-275, Vol. 3 No. 8 from a printed publication issued and published in the "Journal of the Aeronautical Sciences" about June, 1936, and that said copy may be used in evidence with the same force and effect as an original, subject to any objection which may be made thereto as irrelevant or immaterial when offered in evidence, viz.:

LYON & LYON,
/s/ FREDERICK W. LYON,
Attorneys for Plaintiffs.

/s/ FRED GERLACH,
/s/ ROBERT B. WATTS,
Attorneys for Defendants.

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JOURNAL OF THE AERONAUTICAL SCIENCES

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Technological Developments of the Curtiss-Wright "Coupe"

Presented by T. P. Wright at the Pacific Coast Meeting of the I. Ae. S., February 7, 1936

ALBERT E. LOMBARD, Jr., *Curtiss-Wright Airplane Company*

Summary

THIS paper presents the results of research which was carried out in the development of the Curtiss-Wright "Coupe," a two place, all-metal cantilever monoplane. Wind tunnel data of the effects of split flaps is reported, as is also that dealing with the drag of certain features of the airplane. Structural tests of a series of stiffened sheet metal panels in edge compression are reported which show good correlation with the "effective width" conception of the action of thin sheet in the buckled state. Comparison is made of fifteen types of stiffeners suitable for use on reinforced sheet structures subjected to compression. The results of flight tests and theoretical studies combined with wind tunnel tests of airfoils are discussed, which indicate that the stalling characteristics of tapered monoplane wings can be appreciably improved without the use of aerodynamic twist, by using a highly cambered airfoil at the tip having a high value of $C_{L\,max}$.

Aerodynamic Design

Wind tunnel tests were conducted on a 1/12 scale model of the preliminary design in the Buffalo wind tunnel of Curtiss Aeroplane and Motor Company, shown in Fig. 1. These tests included the effect on the lift and pitching moments of the installation of split flaps, 20% of the wing chord, 42% of the span, set at 60 degrees, (Fig. 2). It is seen that the flaps produced a positive pitching moment (tail heavy) on the complete model, which is desirable since thereby the airplane can be glided at a reduced airspeed after the flaps have

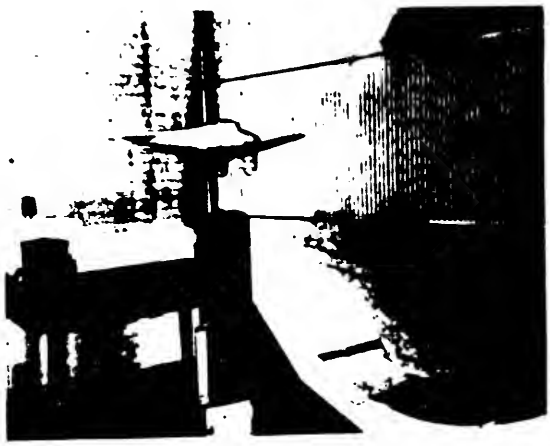

Fig. 1. Model in Buffalo wind tunnel of Curtiss Aeroplane and Motor Company.

been extended without retrimming the elevator. This effect has been checked in flight test and found to simplify controlling the glide to the field. With flaps down, the speed of trim is about 10 m.p.h. less than with the flaps neutral, so that a constant margin is maintained over the stalling speed which is also reduced 10 m.p.h. by the flaps. Fig. 2 further shows that the middle portion of the flap is more effective in producing this positive moment than in increasing the maximum lift coefficient.

The results of certain of the drag tests run at 60 m.p.h. are shown in Figs. 3 and 4. Interpreted in the terms of the two units, "percentage of total airplane drag," and "miles per hour," these items become:

EXHIBIT 18

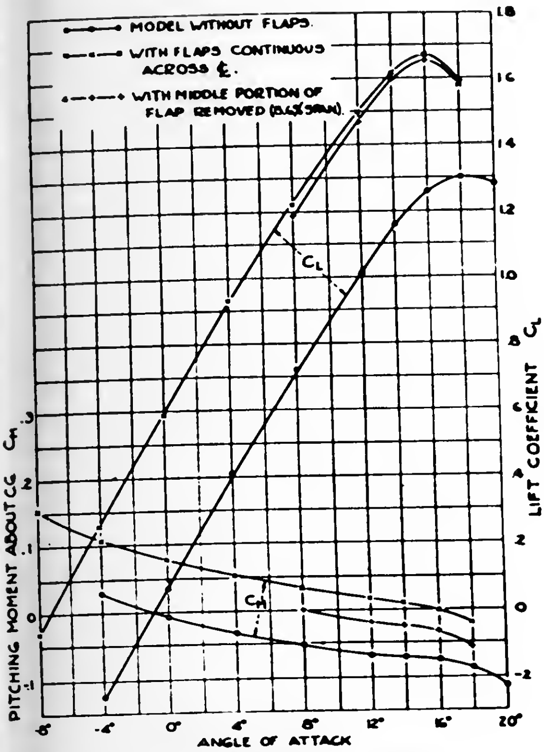

FIG. 2 EFFECTS OF FLAPS.

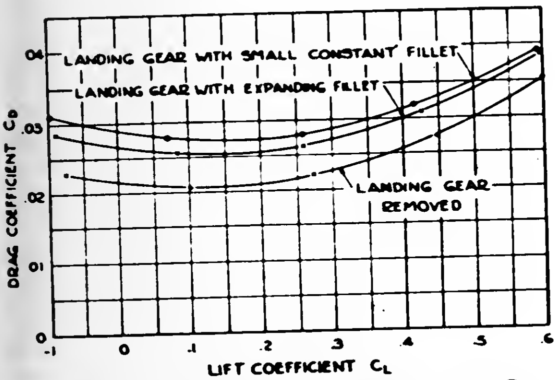

FIG. 3. DRAG OF LANDING GEAR.

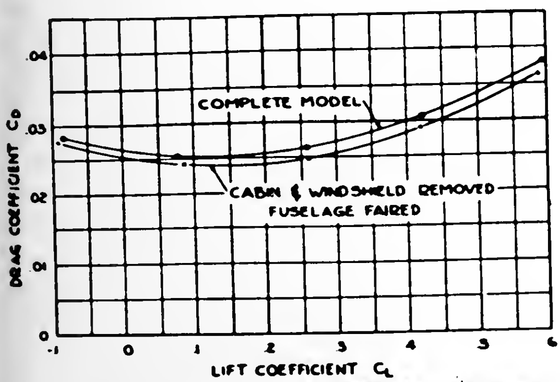

FIG. 4. DRAG OF CABIN & WINDSHIELD.

TABLE I TESTS OF STIFFENED 24ST ALCLAD PANELS

Stiffener Type	Material	Stiffener Area sq in	Panel Thickness in	No. of Stiffeners	Load Carried lbs	Total Effective Area sq in	Effective Stress lb/sq in	Average Effective Stress lb/sq in
1	24ST Alclad	.0617	.0315	2	5560	.269	20,600	20,000
		.0606	.0317	2	5270	.272	19,400	
2	24ST Alclad	.0753	.0321	2	7490	.289	25,900	27,000
		.0755	.0313	2	7870	.280	28,100	
3	24ST Alclad	.0794	.0320	2	7490	.297	25,200	24,500
		.0817	.0318	2	7160	.301	23,700	
4	24ST Alclad	.0777	.0099	2	9890	.344	28,700	27,000
		.0771	.0318	2	7500	.289	25,300	
5	24ST Alclad	.0944	.0317	2	8060	.329	24,500	24,000
		.1034	.0320	2	8210	.331	23,400	
6	24ST Alclad	.0773	.0318	2	9790	.276	35,500	34,500
		.0774	.0320	2	9800	.279	34,200	
7	24ST Alclad	.0764	.0317	2	8640	.280	30,800	31,800
		.0757	.0317	2	8140	.277	31,600	
		.0745	.0316	1	5650	.175	31,600	
		.0752	.0320	1	6040	.180	33,000	
8	24ST Alclad	.0764	.0326	2	8060	.291	27,700	28,600
		.0763	.0316	2	8330	.283	29,500	
9	17ST Tube	.1009	.0319	2	9450	.326	29,000	28,900
		.0961	.0315	2	9310	.323	28,800	
10	17ST Tube	.0705*	.0319	2	8830	.287*	30,800*	30,400*
		.0705*	.0322	2	8490	.291*	29,500*	
11	24ST Extruded	.0882	.0323	1	6630	.175	36,600	32,400
		.0888	.0324	1	6310	.175	33,100	
		.0880	.0316	2	9400	.306	30,700	
		.0880	.0315	2	10170	.302	33,700	
		.0886	.0318	2	9790	.309	32,100	
		.0882	.0322	2	10210	.306	33,600	
		.0870	.0342	2	9140	.292	31,400	
		.0870	.0306	2	9330	.289	32,400	
		.0878	.0322	3	11590	.420	27,600	
		.0886	.0320	3	12060	.422	28,600	
		.0870	.0340	4	16090	.501	32,100	
		.0870	.0306	4	16280	.498	32,700	
		.0880	.0265	1	5640	.167	33,800	
		.0888	.0272	1	5830	.169	34,600	
		.0882	.0274	2	9730	.277	34,100	
		.0880	.0275	2	9660	.278	34,700	
		.0875	.0208	1	4440	.144	30,400	
		.0870	.0207	1	4640	.143	32,400	
12	24ST Extruded	.104	.0313	2	11020	.328	33,600	34,200
		.104	.0312	2	11300	.326	34,100	
13	24ST Extruded	.148	.0309	2	16280	.448	36,400	36,800
		.148	.0306	2	16560	.444	37,200	
14	24ST Extruded	.0756*	.0316	2	8500	.281*	30,200*	30,200*
		.0757*	.0324	2	8150	.284*	30,300*	
15	24ST Extruded	.172	.0305	2	15840	.481	32,900	32,200
		.172	.0304	2	15980	.483	31,500	

* AVERAGE AREA USED (= VOLUME/LENGTH)

	Percentage of Total Drag	Speed Effect
Landing gear drag:		
With constant fillet.........	18%	7 m.p.h.
With expanding fillet.......	13%	5 m.p.h.
Reduction due to expanding fillet	5%	2 m.p.h.
Cabin and windshield drag: (compared to smoothly faired fuselage, cabin removed)..	5%	2 m.p.h.

Structural Design

Edge compression tests of sheet metal panels of 24ST Alclad stiffened by various types of formed and extruded shapes were carried out to check the "effective width" method for computing the strength of stiffened sheet metal panels, and to select a suitable stiffener type.

The panel tests indicate that the "effective width" method, developed by von Karman, Sechler and Donnell,[1,2] and by Lundquist[3] (Method C), which assumes the stiffener and an effective width of adjacent sheet to act as a unit, all at the same stress, gives good coordination of the results. Referring to Table 1, it is seen that the "effective stresses" for stiffeners Type 7 and Type 11, which were tested on panels of various gauges with various numbers of stiffeners, lie within narrow bands which are no broader than the individual variations on supposedly identical panels. One exception to this close correlation occurred in the two tests of stiffener Type 11 with .032 sheet and three stiffeners per panel in which the sheet failed prematurely due, apparently, to a peculiarity of the rivet pattern on those panels, but this exception is not believed to invalidate the rule established. The "effective width" method of analysis is considered very satisfactory.

Stiffener Selection

In selecting a suitable stiffener type, certain restrictions were necessarily placed on the type of stiffener and its method of attachment to the sheet. These restrictions were:

(1) The stiffener should be of a type that attaches to the wing skin with one row of ⅛ inch dural modified brazier head rivets spaced 1 inch apart.

(2) The stiffener should be of a type suitable for use where the rib spacing would be approximately 20 inches.

(3) The stiffener should have sufficient area and strength such that, when used with .032 inch thick 24ST Alclad, the stiffener spacing at the root of the wings need be not less than 4 inches. This third requirement was only partially adhered to.

The tests are summarized in Table 1. It was endeavored to test representative sections of all types, some obviously designed for ease of fabrication, some for high structural efficiency. Inasmuch as any stiffener could be made somewhat larger or somewhat smaller if the area was not consistent with the load to be carried, the final selection of the stiffener type was based on the maximum effective stress which the stiffener was able to carry.

The eligible types for general use were Types 1 to 12. Types 13 to 15 were intended for use in special places and on other models involving higher loads. Of the "eligible" types, the formed-up stiffener, Type 6, was the strongest. This stiffener had several desirable characteristics in its shape and design which are worthy of comment:

(1) This stiffener had sufficient depth for the length tested to prevent failure by bowing as an Euler column.

(2) This stiffener was made of sufficiently thick material that it did not buckle locally. This characteristic was made possible by the fact that the stiffener had a small developed width. It can be generally concluded that the developed width of stiffeners should be as small as is consistent with the desired depth for Euler strength.

(3) The formed bulb of this stiffener was so shaped that the stiffener was well supported laterally without unduly stressing the free edge to cause it to roll out flat.

(4) The distance from the vertical leg of the stiffener to the line of rivets was small, thus enabling the vertical leg to give considerable support to the sheet panel.

(5) The upturned roll on the free edge of the riveted leg offered support to this leg and to the attached sheet.

The extruded stiffeners, Types 11 and 12, which were next best in order of merit, were selected for actual use because of:

(1) The high maximum stress which they developed,

(2) The manner in which they failed without sudden collapse so that after failure they were still able to carry a large percentage of their maximum load,

(3) The uniformity of manufacture of the extruded shapes, and

(4) The low fabricated cost of the extruded shapes.

It should be noted in passing that the extruded shapes were made of 24ST aluminum alloy, while the formed-up stiffeners, Type 6 specifically, were made of 24ST Alclad which is approximately 10% weaker than straight 24ST. Presumably, if the formed-up stiffener, Type 6, were made of pure 24ST it would be considerably stronger than the extruded shapes, but corrosion difficulties prevent the use of unprotected 24ST in thin sheets.

Test Equipment

All the sheet metal panels were supported in the framework, Fig. 5. The load was applied to each specimen by blocks which clamped on the end of the sheet. The stiffeners, which butted against the steel face plates, were finished on the ends and accurately attached to the sheets to provide the correct

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[1] Th. von Karman, E. E. Sechler, and Donnell, *The Strength of Thin Plates in Compression,* Applied Mechanics Transactions A.S.M.E., June, 1932.

[2] E. E. Sechler, *The Ultimate Compressive Strength of Thin Sheet Metal Panels,* Thesis at Calif. Inst. of Tech. 1934.

[3] Eugene E. Lundquist, *Comparison of Three Methods for Calculating the Compressive Strength of Flat and Slightly Curved Sheet and Stiffener Combinations,* NACA Tech. Note No. 455, 1933.

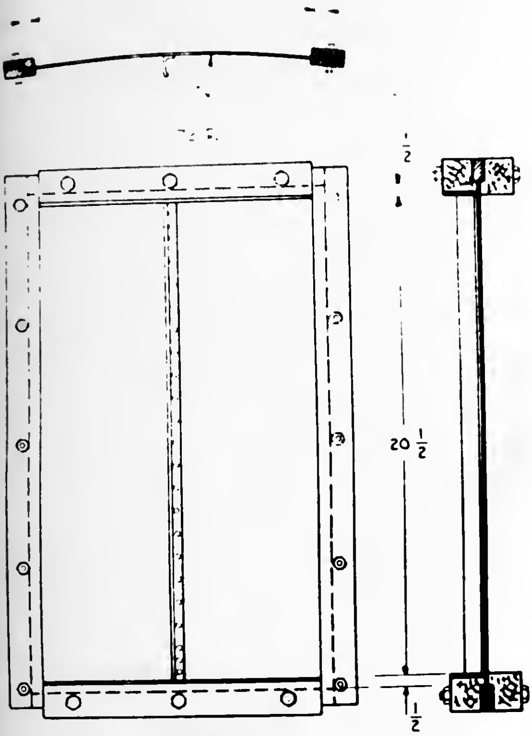


FIG. 5 TEST JIG FRAME

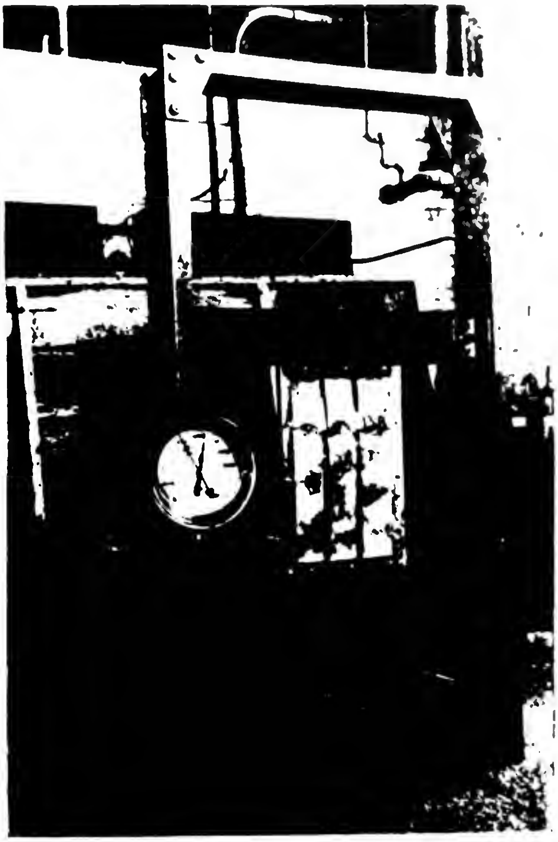


FIG. 6. Hydraulic test machine for sheet metal panels.

overhang of the sheet on the end to fit the depth of the clamping blocks. In order to take care of small variations in the ends of the stiffeners, small brass shims (.002 in. to .010 in. thick) were inserted until they were tight before the load was applied.

The vertical guides of this frame were made of steel and held to the 12 inch spacing by steel bars across the back. The thickness of the slot thru which the sheet could slide was accurately maintained by clamping shims between the bars slightly thicker than the sheet to be tested.

This method of supporting the edges of the sheet was found to be very satisfactory when testing panels with stiffeners. On such panels the buckling was always more severe in the middle of the panels, and the failures always occurred in the middle of the panels—never along the edges—at the instant when the stiffeners failed. The frictional load carried in the guides, up to the point of failure of the stiffeners, could be only a relatively small percentage of the total load in the sheet edges. Previous to the time of failure the guides would drop freely of their own weight whenever the load was removed. It is estimated that the frictional load in the guides increased the observed maximum load by possibly 50 pounds, which is considered negligible.

The testing machine shown in the photograph, Fig. 6, developed after the fashion of the one described, (see Reference 2), incorporates a hydraulic jack with a maximum load capacity of 20,000 pounds.

Method Used to Compute Stress in Stiffener and Sheet in Panel Tests

Reference 2 shows that the load carried by a simply supported, unstiffened sheet metal panel can be represented by the formula

$$P = Ct^2\sqrt{E\sigma},$$

and the "effective width" can be written

$$2w = Ct\sqrt{E/\sigma},$$

in which

C is a function of η and λ as shown in Fig. 9.

$$\eta = (b/R)\sqrt{E/\sigma}$$

$$\lambda = (t/b)\sqrt{E/\sigma}$$

E is Young's modulus of elasticity.

σ is stress at the supported edge of a panel and also, in the case of stiffened panels, the stress in the stiffener and adjacent sheet (of effective width = 2 w).

t is thickness of the sheet.

b is width of sheet panels between supports.

R is radius of curvature of sheet.

The curves of C given in Fig. 9 for the range of η and λ encountered in the wings were derived from data

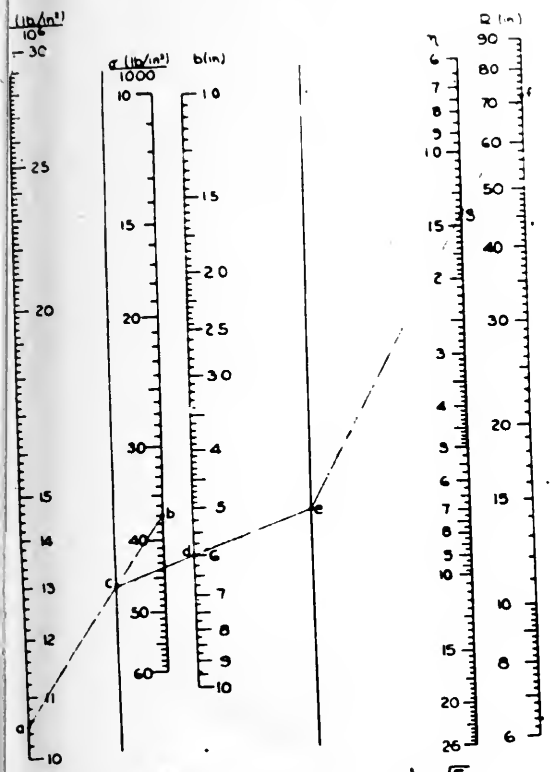


FIG. 7 SOLUTION OF $\eta = \frac{b}{R}\sqrt{\frac{E}{\sigma}}$

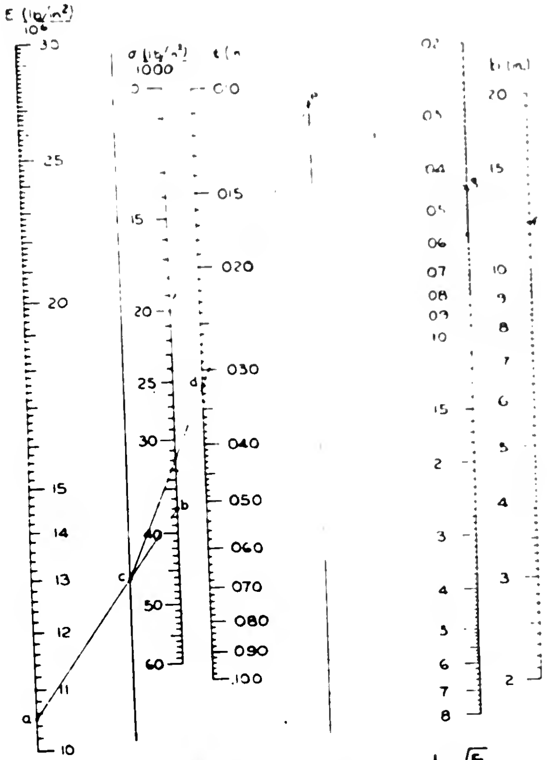


FIG. 8. SOLUTION OF $\lambda = \frac{t}{b}\sqrt{\frac{E}{\sigma}}$

of Reference 2. The nomograms, Figs. 7, 8, 10 and 11, were developed to simplify the computation of the parameters and the values of $2w$ and P.

The assumed stress distribution in the stiffeners and sheet of the panel tests is shown in Fig. 12, in which the stresses in the stiffeners and adjacent sheet elements and in the edges of the sheet are all the same inasmuch as the shortening of the panel under load is the same at all points. Because of the support offered the sheet by the side guides, it was assumed also that all the width of the sheet inside the guides would act effectively at the maximum stress. The justification of this assumed stress distribution is believed to lie in the close correlation of the tests of the stiffeners Types 7 and 11, Table 1.

Wing Static Test

A complete wing for one side was static tested to the full design loads for high angle of attack and inverted flight. This wing, designed to close margin in accordance with the method as developed in the panel tests, carried the design loads without failure. Clips and fittings, designed according to this assumed distribution of stress in the sheet and stiffener, were all found satisfactory. These facts speak for the practical applicability of the method of analysis to the design of all-metal aircraft.

Flight Tests

The most interesting part of the flight test program was that devoted to the stalling characteristics in which modifications were effected in the wing contour which enabled the airplane to be stalled in a smooth and controllable manner.

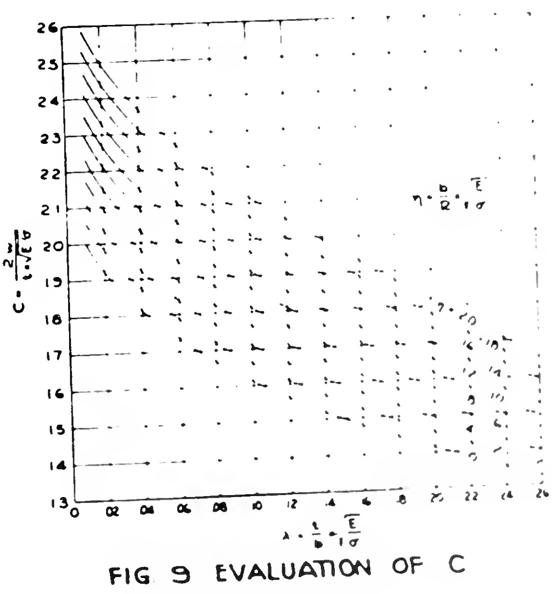


FIG 9 EVALUATION OF C

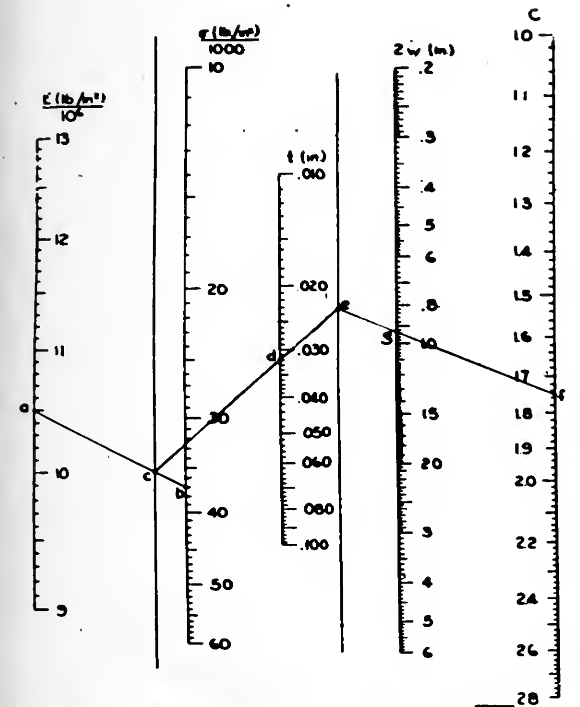


FIG. 10. SOLUTION OF $2w = Ct\sqrt{E/\sigma}$

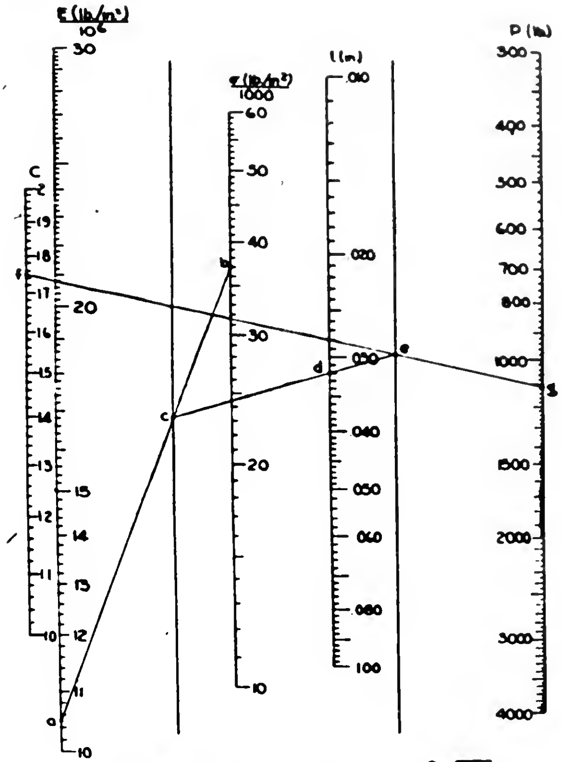


FIG. 11. SOLUTION OF $P = Ct^2\sqrt{E\sigma}$

As originally flown, the airplane had a straight tapered wing with an N.A.C.A. 2315 airfoil at the root and N.A.C.A. 2309 airfoil at the tip with no twist. The N.A.C.A. 2309–2312–2315 series was selected because, on the average, it showed the smoothest shaped lift curve peaks of all the low cambered, low drag airfoils tested in the N.A.C.A. Variable Density Tunnel.[4] The stall of this wing was observed in flight, by wool tufts, to start at the leading edge near the right wing tip and progress rapidly to cover the whole tip portion of that wing, whereupon it would drop uncontrollably. The conditions with the split flaps extended were essentially the same as with them retracted.

Flight tests were then carried out with fixed auxiliary airfoils, 14.5% chord, extending over the outer 50% of the span. Two types were investigated, one with a symmetrical N.A.C.A. 0012 section and the other with a highly cambered N.A.C.A. 22 section.[5]

It was found that, under certain combinations of angles, these fixed auxiliaries improved the stalling characteristics by reducing the autorotational tendencies and improving the aileron control. The effects of these auxiliaries were quite insensitive to their angular setting; i.e., a large change in angular setting was necessary to bring about an appreciable change in the stall.

The cambered auxiliaries appeared to be better than the symmetrical in their effects on the stall. However, the installation of either type of auxiliary was so detrimental to the take-off and climb characteristics, particularly with the auxiliaries at the angles necessary for the best stall characteristics, that the use of the fixed auxiliaries could not be considered satisfactory and was, therefore, abandoned.

The other course which was followed to improve the stalling characteristics was to modify the airfoil sections on the outer portions of the wing by fairing out the under side of the leading edge in successive steps, increasing the leading edge radius, and increasing the airfoil camber. This procedure was found definitely to improve the stalling characteristics. With the final configuration, the CW-19 airfoil (Fig. 13) at the tip tapered to the N.A.C.A. 2315 airfoil at the root, all autorotational tendencies below the stall were eliminated and the airplane could be positively controlled in the stalled condition. The wool tufts showed that the stall of this wing started along the trailing edge near the mid point of the semi-span and proceeded gradually in all directions. The leading edge at the tip remained unstalled throughout. It is interesting to note that when the nature of the stall was changed so that the separation started at the trailing edge, instead of at the leading edge, the whole character of the stall became smooth, more controllable.

It appeared that the change to the CW-19 airfoil at the tip was equal in effectiveness at the stall to the

[4] Jacobs, Ward and Pinkerton, *The Characteristics of 78 Related Airfoil Sections from Tests in the Variable Density Wind Tunnel*, N.A.C.A. Tech. Report 460, 1933.

[5] Fred E. Weick & Robert Sanders, *Wind Tunnel Tests on Combinations of a Wing with Fixed Auxiliary Surfaces Having Various Chords and Profiles*, N.A.C.A. Tech. Report 472, 1933.

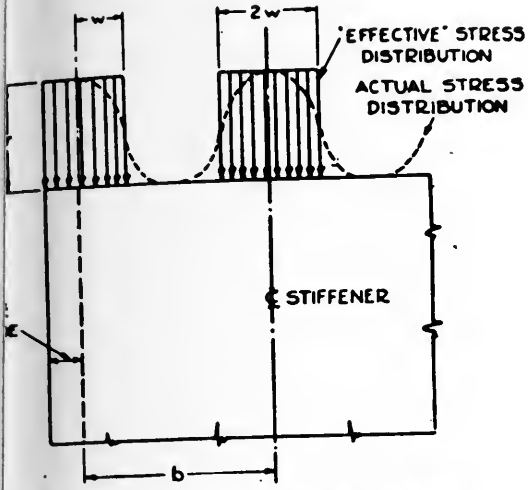


FIG.12. STRESS DISTRIBUTION.

stallation of either type of fixed auxiliary airfoil. There was no observable adverse effect on the stability or performance due to this modification.

Theoretical Investigation of Tapered Twisted Wings

One method of improving stalling characteristics of a wing is to use aerodynamic twist reducing the incidence along the span so that the tip will stall at a higher angle of attack than the root. The effects of this twist can be determined analytically by the method developed by Glauert.[6] (Chapter XI.) The circulation about any point on the wing span is expressed by the Fourier series

$$\Gamma = 2bV\Sigma A_n \sin n\theta \qquad (1)$$

where

Γ is circulation ($= C_L c V/2$)

b is wing span

c is wing chord at any point

V is velocity at infinite distance from wing.

θ represents point on wing span defined by the equation:

$y = (b/2)\cos\theta$

y is distance out from center line.

The downwash velocity w at any point θ becomes

$$w = V\Sigma nA_n \sin n\theta / \sin\theta \qquad (2)$$

By equating the circulation defined by the basic series (Eq. 1) to the circulation derived from the angle of attack as affected by downwash (Eq. 2), and assuming a straight line variation of lift coefficient with angle of attack, the following equation is obtained for the tapered wing with constant twist along the span:

$$\Sigma A_n \sin n\theta[n/\sin\theta + 1/\mu] = \bar{\alpha} - (\cos\theta)\epsilon \qquad (3)$$

where

$\mu = (c/4b)(dC_L/d\alpha_0)$

$\bar{\alpha}$ is absolute angle of attack at the root measured from zero lift

ϵ is aerodynamic twist at tip (positive when the angle of attack is less at the tip than at root.)

The first four coefficients of the series A_1, A_3, A_5 and A_7 can be evaluated by satisfying Eq. (3) at the four points $\theta = 22\frac{1}{2}°, 45°, 67\frac{1}{2}°, 90°$. This evaluation has been made for a straight tapered wing as follows:

Aspect Ratio $R = b^2/S = 6.724$

Tip Chord/Root Chord $= .464$

$dC_L/d\alpha_0 = 5.84$ radian

$$A_1 = .21543\,\alpha - .08832\,\epsilon \qquad (4)$$
$$A_3 = .00474\,\alpha - .01585\,\epsilon$$
$$A_5 = .00063\,\alpha + .00362\,\epsilon$$
$$A_7 = .00087\,\alpha - .00613\,\epsilon$$

The total lift of an airfoil is (by page 136, Ref. 6)

$$L = \frac{1}{2}\pi b^2\rho V^2 A_1$$

when the average lift coefficient is

$$C_{Lav} = 2L/S\rho V^2 = \pi b^2 A_1/S = \pi R A_1 \qquad (5)$$

It is now convenient to evaluate the coefficients in terms of C_{Lav} and ϵ, which become

$$A_1 = .01734\,C_{L} \qquad (6)$$
$$A_3 = .00104\,C_L - .01424\,\epsilon$$
$$A_5 = .00212\,C_L + .00757\,\epsilon$$
$$A_7 = .00040\,C_{La} - .00640\,\epsilon$$

The induced drag [illegible] (by page 136, Ref. 6)

$$D_i = \frac{1}{2}\pi b^2\rho V^2\Sigma nA_n^2$$

[illegible]

$$C_{Di} = \pi R\Sigma nA_n^2 \qquad (7)$$

[illegible]

$$C_{Di} = .01788\,C_L^2 - [illegible]\,C_L\epsilon + [illegible].1333\,\epsilon^2$$

[illegible] C_{Di} [illegible] C_L [illegible] πR

[illegible]

$$C_{Di} = .00534\,C_L^2 - [illegible]\,C_L\epsilon + [illegible].1333\,\epsilon^2 \qquad (9)$$

[illegible]

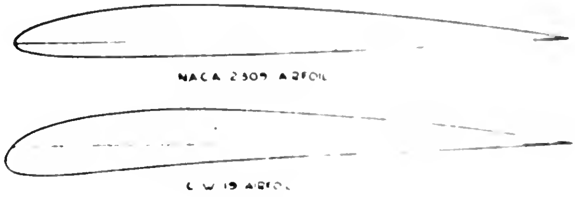


FIG 13

[6] Glauert, *Airfoil and Airscrew Theory*, Cambridge Press, London.

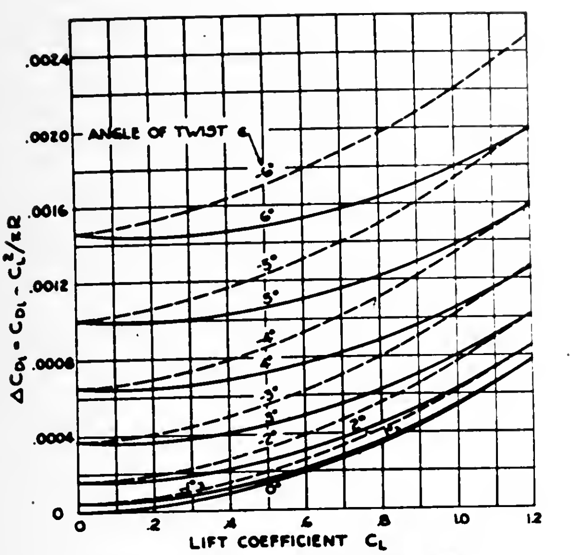


FIG. 14. ΔC_{D_i} FOR TAPERED WING.
ASPECT RATIO = 6.724 $dC_L/d\alpha_0$ = 5.84/rad.
TIP CHORD = .461 ROOT CHORD

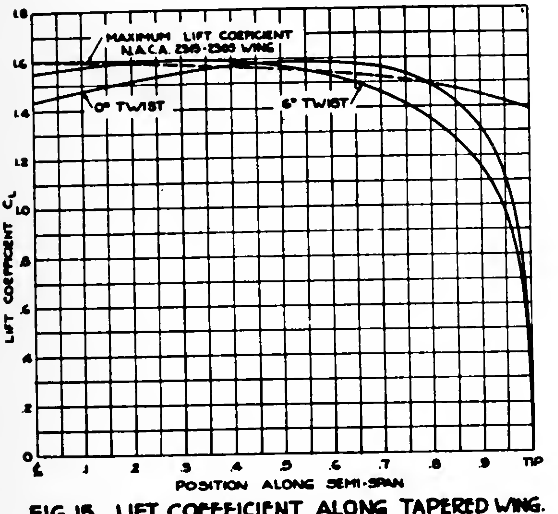


FIG. 15. LIFT COEFFICIENT ALONG TAPERED WING.
$C_{L_{AV.}}$ = 1.50
ASPECT RATIO = 6.724 TIP CHORD = .461 ROOT CHORD.

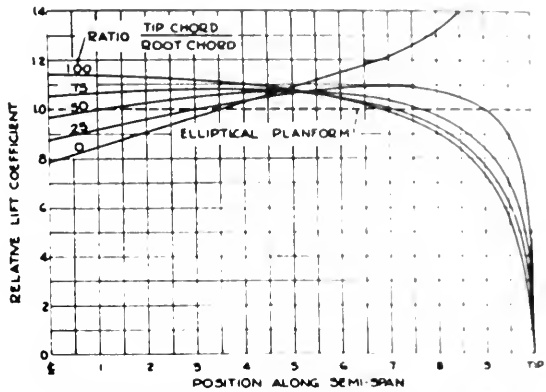


FIG. 16 LIFT COEFFICIENT GRADING CURVES
TAPERED WINGS A R = $dC_L/d\alpha_0$ = 5.8

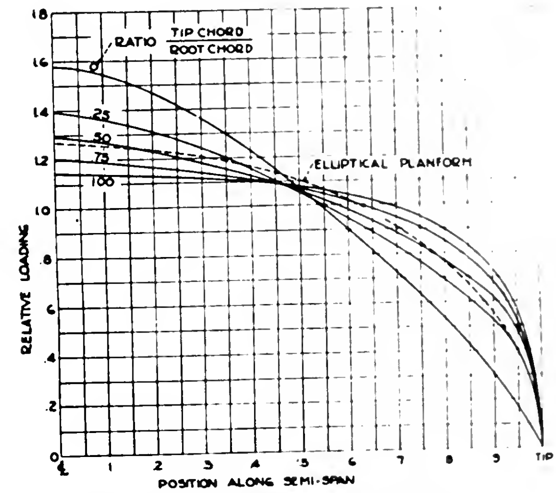


FIG. 17. LOAD GRADING CURVES
TAPERED WINGS A R = $dC_L/d\alpha_0$ = 5.8

Curves are plotted in Fig. 14 giving the values of ΔC_{Di} for various angles of twist. For a twist up to 2° the induced drag is not serious, amounting to not over 1% of the drag of an average airplane, but as the twist is increased above 2° the drag becomes appreciable.

The lift coefficient at any one of the four points $\theta = 22\frac{1}{2}°$, 45°, $67\frac{1}{2}°$ and 90° is obtained from Eq. (1) to be:

$$C_L = 2\Gamma / c\,V = (4\,b/c)\,\Sigma A_n \sin n\,\theta \tag{10}$$

In Fig. 15 are plotted the lift coefficients along the span of this wing with zero twist and a wing with a hypothetical 6° of twist, both at $C_{L_{av.}} = 1.50$. In this figure is plotted also a curve for the maximum lift coefficient along the span which was developed taking into account the variation in the maximum lift with airfoil thickness ratio and with Reynolds Number, which varies along the span due to the taper.

It is seen that the wing with 0° twist exceeds the maximum lift coefficient for the 2315–2309 series over a considerable portion of the outer wing, and it is therefore reasonable that there should be a pronounced tendency to stall at the tip first, bringing about uncontrollable autorotation. The curve with 6° twist represents a wing that should be satisfactory in the stall if the N.A.C.A. 2315 to 2309 wing were retained. However, referring again to Fig. 14 it is seen that such a wing would have an appreciably higher drag than the untwisted one. It is to be concluded, therefore, that to try to obtain good stalling characteristics merely by twisting the wing is decidedly inefficient. It is much better to use only 1°–2° of aerodynamic twist in combination with a tip airfoil having a high value of $C_{L\,max}$ and having a lift curve with a round smooth top.

It is of interest to note that the benefits gained by substituting the CW-19 airfoil for the N.A.C.A. 2309

tip were due entirely to the extension of to a high value of C_{Lmax} at a high angle There was only 0.2° shift of the zero lift the N.A.C.A. 2309 airfoil to the CW-19 red to a common chord as determined by Buffalo wind tunnel of the Curtiss Aero-Motor Company. However, these tests on rfoils at 80 m.p.h. showed that the CW-19 oped a high uncorrected $C_{Lmax.} = 1.36$ with unded lift curve peak, comparable to $C_{lmax.}$ $= 1.00$ for the N.A.C.A. 2309 and $C_{lmax} = 1.18$ for the Clark Y.

Figs. 16 and 17 have been prepared using coefficients from Reference 6 to show the load grading curves and lift coefficient grading curves for a series of airfoils with various taper ratios for an aspect ratio $R = d\,C_L/d\alpha_0 =$ approximately 5.8. It is important to recognize that while structural efficiency is gained with the high taper ratios, the problems of obtaining good stalling characteristics are increased.

Admitted November 24, 1950.

DEFENDANTS' EXHIBIT WW

District Court of the United States, Southern District of California, Central Division

Civil Action No. 10930-Y

MAURICE A. GARBELL, INC., a California Corporation, and GARBELL RESEARCH FOUNDATION, a California Corporation,

Plaintiffs,

vs.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION, a Delaware Corporation, and AMERICAN AIR LINES, INC., a Delaware Corporation,

Defendants.

STIPULATION #3

It is hereby stipulated subject to proof of error that the appended are reproductions of the following printed publications and that the said copies may be used in evidence with the same force and effect as originals, subject to any objection which may be made thereto as irrelevant or immaterial, when offered in evidence, viz:

"Exhibit 19" is a reproduction from a printed publication, Vol. XLI, pages 175-180, entitled "Aerodynamic and Structural Features of Tapered Wings" issued and published during the year 1937; by the "Royal Aeronautical Society" of London, England.

"Exhibit 20" is a copy of a reproduction of a publication entitled "Correspondence," Vol. XLII, pages 754-755, issued and published during the year 1938 by the "Royal Aeronautical Society" of London, England.

"Exhibit 21" is a reproduction of pages 660, 661, 671, 672, 690, and 697, Vol. XXII, of an article entitled "Development of Sailplanes" issued and published during the year 1938 by the "Royal Aeronautical Society" of London, England.

LYON & LYON,
/s/ FREDERICK W. LYON,
Attorneys for Plaintiffs.

/s/ ROBERT B. WATTS,
/s/ FRED GERLACH,
Attorneys for Defendants.

The better response to ailerons and its resulting effect of manœuvrability which is afforded by wings of higher taper ratio can therefore only be utilised ...re is taken to maintain a sufficient degree of lateral control at and beyond ... stall.

THE STALLING OF TAPERED WINGS.

This subject has recently received a good deal of attention in this country and America in view of the unpleasant characteristic of tapered wings, especially ...se of high taper ratios, of dropping a wing when stalled in a more vicious way ...an rectangular wings. It has also been observed in flight and on models in the ...nd tunnel that for highly tapered wings there is a very definite tendency to stall ...st at the tips and not at the centre. The stalling characteristics of wings of ...w taper ratios are still very much disputed, and some designers of aircraft ...ing wings of relatively small taper ratio claim stalling characteristics comparable to those of rectangular wings.

When first faced with the phenomenon one is inclined to explain the behaviour ... the stall of tapered wings solely on the basis of the aerofoil theory. The ...rofoil theory indicates, as illustrated in Fig. 7, that an elliptical wing or a wing

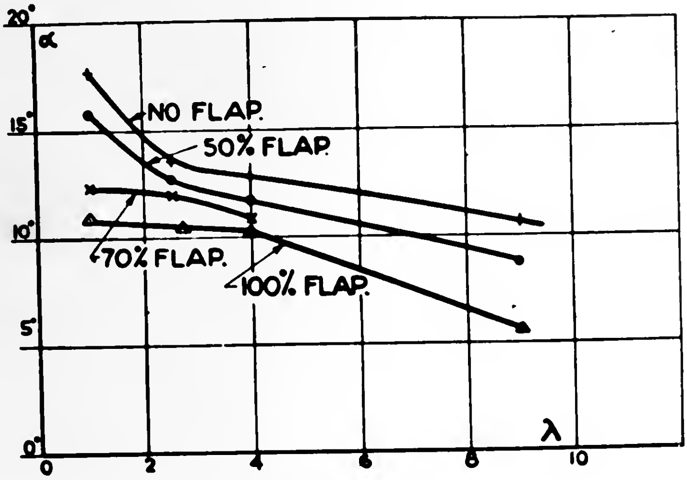


FIG. 10.

... taper ratio of about 2 : 1 which approaches the elliptical distribution should ...all simultaneously over the whole span. Wings of higher taper ratio should ...all first at a point somewhat inboard of the wing tips as there the local C_L or ... effective angle of incidence reaches a maximum value prior to other portions ... the wing. However, it has been found that the aerofoil theory alone does ... give a satisfactory explanation and that a number of other parameters have ... be considered. Tests carried out by Millikan (14) at the Pasadena Institute ... Technology, indicated that for a wing of a given taper ratio the characteristics ... stalling changed decisively as the aspect ratio of the wing was increased.

More recent tests by Irving at the N.P.L. and observations in flight by ...ray (15) have indicated the existence of a spanwise flow which depends on the ...ection of sweepback. On a tapered wing with no sweepback of the leading ...ge and a sweep forward of the trailing edge, Irving observed a transverse ...ow near the trailing edge which was directed from the tips towards the centre ... the aerofoil. A similar type of flow was observed by Gray on wings which ... negative angle of yaw. *Vice-versa* an outward flow (towards the tips) ...s observed on a tapered wing having a swept back leading and correspondingly ... full-scale on a monoplane with positive angle of yaw. Corresponding to the ...ection of this secondary flow the stalling of the tips was either delayed when

the flow had an inward direction and accelerated when the flow had an o direction. The explanation for this phenomenon, as given by Gray (*F* July 16th, 1936), due to the transverse pressure gradient, is not conv C. N. H. Locke, of the N.P.L., has pointed out in a letter addressed to *Flig* (August 27th, 1936) that in the case of a yawed aerofoil the flow may be re into a two-dimensional flow in planes normal to the aerofoil together w uniform velocity along the span which will not affect the equilibrium of the verse flow. The spanwise component of the flow will affect the boundary especially when the aerofoil is stalled. In the case of a yawed aerofoil, the case of an aerofoil with swept forward trailing edge, dead air will be ported from the tips towards the centre thus delaying the stalling of the tips accelerating the stalling of the tips and accelerating the stalling of the cen comparison with the corresponding aerofoil with straight trailing edge.

This aspect of stalling still requires fuller research, and it seems a littl early to form a definite opinion, but it is most likely that the phenomenon o spanwise dead air transport will explain certain observations in regard t point where the breakaway of the flow first occurs on the wing which a contradiction to the ordinary aerofoil theory.

Apart from this phenomenon it is usually overlooked when applying the ge aerofoil theory that the wing section along the span is not constant on a monoplane wings as the thickness chord ratio varies usually from the c towards the tip, apart from the change in chord.

In predicting the point where stalling will first occur, it is necessary to r allowance for the actual stalling angle of a section at any point of the span. by varying the geometric angle and the characteristics of the section (thic chord ratio and camber) it should be possible to control to some exten commencement of burbling in relation to the wing plan form.

(a) *Influence of Twist.*

A mere twist, *i.e.*, an outwash towards the tips seems to be a very ob scheme to delay the stalling of the tips, but it is, in my opinion, a very inetti way unless the twist becomes so excessively large that the drag and th distribution at small angles of incidence are substantially affected. J. Hueber published some theoretical investigations in 1933 on twisted tapered w The distribution of twist along the span was so chosen as to obtain an ellip C_L distribution. The following table contains the angle of twist and the in of induced drag compared with the minimum value for elliptical lift distrib at an overall $C_L = 1$.

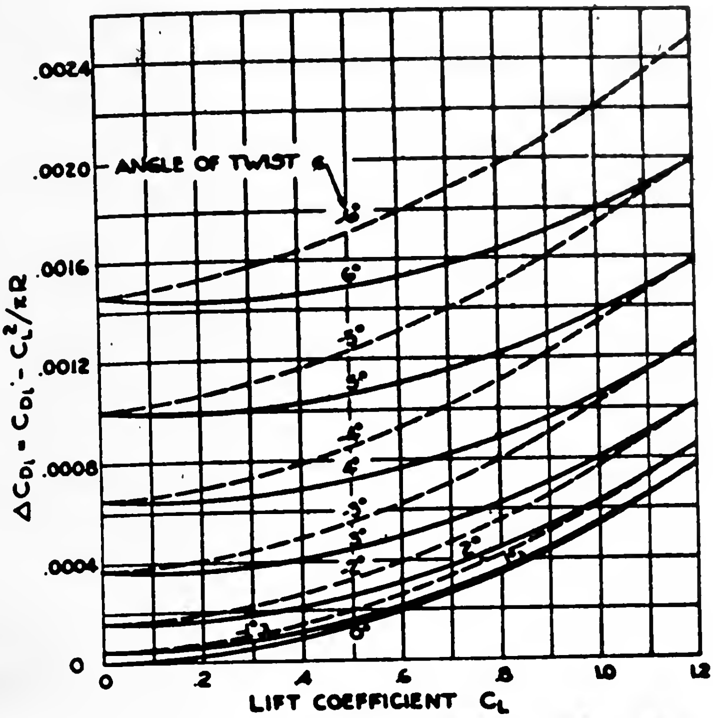
Taper Ratio.	Angle of twist equals difference of geometric angle at root and tip for overall $C_L = 1$.	Di/Di ellip.
5	20	1.21
2.5	19	1.11
1.25	15	1.01
1	13.5	1.0

On a wing which was actually used on a glider consisting of a recta centre portion and tapered tips (taper ratio=1.54) the twist required fo tapered portion was $-9.5°$.

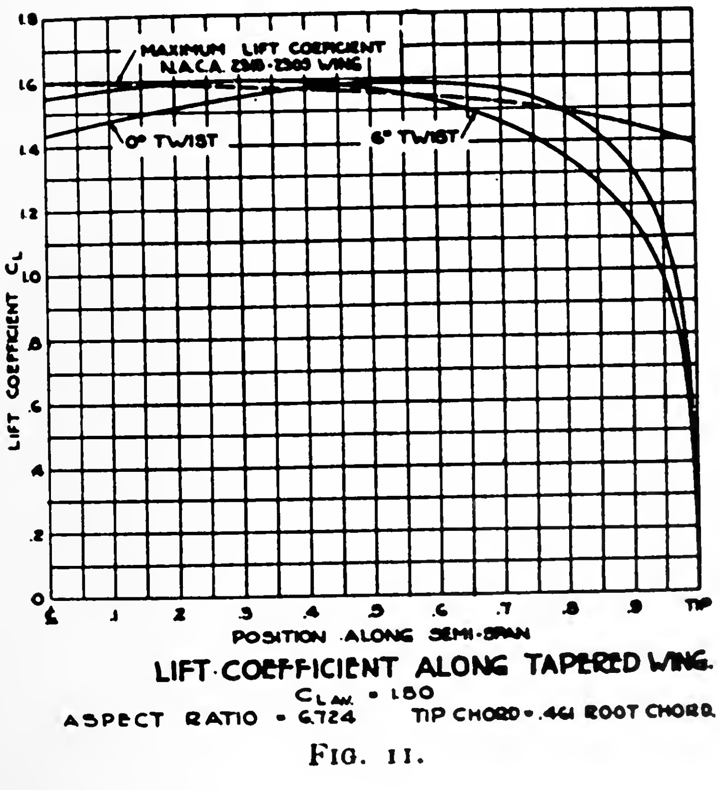
Hueber's assumption of an elliptical C_L distribution, although rationa quite arbitrary and may appear too severe. In a more recent publication o influence of twist by Albert E. Lombard (18) in the *Journal of the Aeron Sciences* ("Technical Developments of the Curtiss Wright Coupé") the a comes to the conclusion that even a mild twist not exceeding $-6°$ is a v inefficient way of obtaining good stalling characteristics. The wing investig by Lombard had an aspect ratio of 6.724 and a taper ratio of 2.16. The in

induced drag for various angles of twist and the resulting distribution are in Fig. 11.

For a twist up to 2° the increase in induced drag is not serious, amounting to not over 1 per cent. of the drag for an average aeroplane, but as the twist is increased above 2° the additional drag becomes appreciable.



ΔC_{Di} FOR TAPERED WING.
ASPECT RATIO = 6.724 $dC_L/d\alpha_0$ = 3.84/rad.
TIP CHORD = .461 ROOT CHORD



LIFT COEFFICIENT ALONG TAPERED WING.
$C_{L\,AV.}$ = 1.50
ASPECT RATIO = 6.724 TIP CHORD = .461 ROOT CHORD

FIG. 11.

(b) Twist Combined with Change of Camber.

More efficient than a mere twist is the combination of twist and chan camber as follows from Fig. 12, where lift curves are plotted for a sect small and a section of higher camber. Provided that the difference in α smaller than the difference in zero lift angle, it is obvious that the total an range for the more highly cambered section is greater than for the section low camber. This increase of total effective angular range can be utilise delay stalling of the tips. If we consider first a section of a relatively low car near the root of the aerofoil, and if we base our consideration on a theoretical C_L distribution depending on the taper ratio of the wing, a cer local value of C_L is required. The margin against stalling of this section

$$\gamma = (\alpha_{max}) \text{ absolute} - \alpha_1 = C_{Lmax}/(dC_L/d\alpha) - C_L/(dC_L/d\alpha)$$

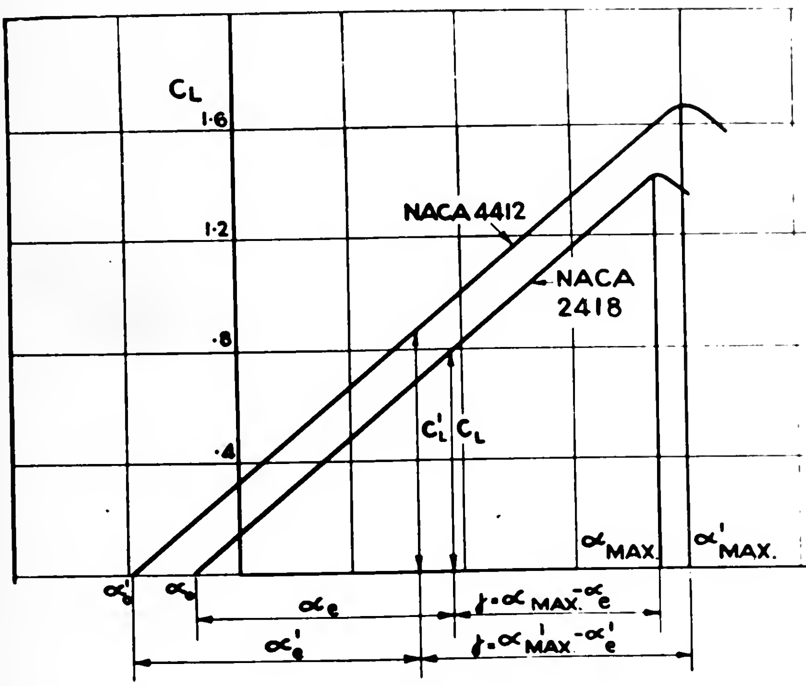


FIG. 12.

$dC_L/d\alpha = 2\pi$ (theoretical value), but this value is actually slightly influenced thickness chord ratio and camber; (α_{max}) absolute $= \alpha_0 + \alpha_{max}$ where α_0 zero angle and α_{max} is the angle at which C_{Lmax} is measured from $\alpha = 0$.

Let us now consider a section further outboard at which the local lift coeffi required may be C_L'. The local margin against stalling at this portion of wing is therefore :—

$$\gamma' = (\alpha'_{max})/\text{absolute} - C_L'/(dC'_L/d\alpha).$$

It is obvious that if $\gamma' > \gamma$, the wing will stall first at the inner section the difference between γ' and γ will then represent the margin against stal of the outboard section compared with the inboard one. It can easily be veri that the required geometric angle and therefore the necessary amount of tw to produce the value of C_L' is equal to the difference of the respective zero angles of the two wing sections.

An investigation on these lines has been made for wings of various tap ratios, and the assumptions in regard to distribution of thickness chord ra

camber ratio along the span are plotted in Fig. 12*a*. This figure contains the amount of twist required in order to produce the theoretical ribution for a given overall C_L value. It seems advisable to choose an all C_L value corresponding to climbing flight. For this condition of flight will then be no increase of induced drag compared with an untwisted wing constant section. Fig. 13 shows the distribution of the margin against ng across the span for wings of various taper ratios and for wing sections ng the maximum camber at various positions of the chord. The characteristics are taken from N.A.C.A. Report No. 460 (1). As the figure indicates ions with the camber at 0.4 and 0.5 of the chord give satisfactory results e sections with the camber at 0.3c are less suitable.

ASSUMED DISTRIBUTION OF THICKNESS & CAMBER ACROSS SEMI-SPAN.

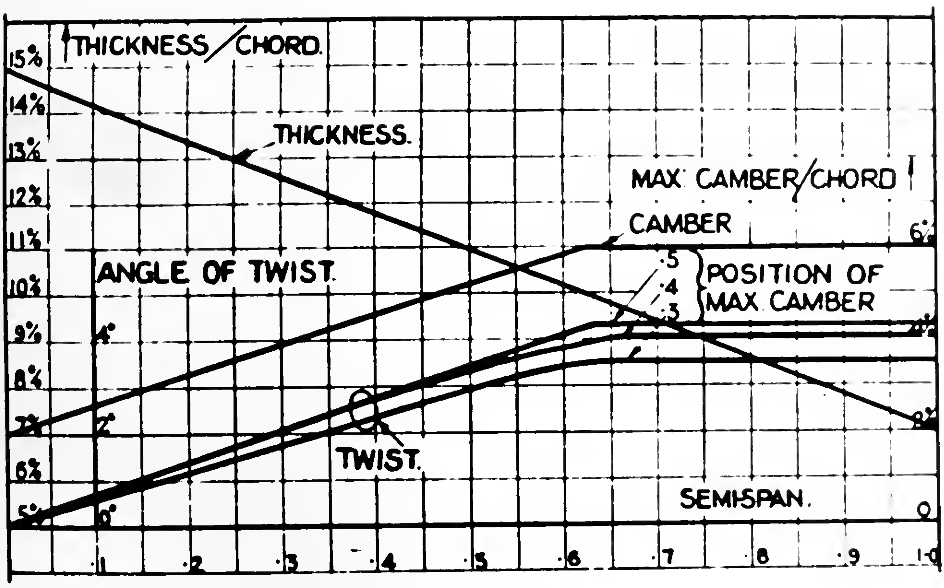


FIG. 12*a*.

Tapered Wings and Wing Tip Slots.

The method described above is based upon the increase of angular range mainly to the lower zero lift angle of higher cambered sections compared with those low camber. The obvious disadvantage, of course, is the difficulty to fair sections of varying camber and also the concentration of high torque at the s where the resistance of the wing against torsional deflection is weakest. other method consists in utilising such sections where the angular range is reased at the high lift end of the angular range, for example, by using a ted section at the tips.

Sucking away the boundary layer is also a means to increase the high lift end the angular range, and one could conceive a method to prevent tip stalling this basis. Such a method would, however, suffer from the obvious practical advantage that the effect is bound up with the working of the power plant ich drives the pump.

5. Tapered Wings and Longitudinal Stability.

(a) *Analysis of Pitching Moments.*

Most designers who began to design monoplanes with tapered w... applied the knowledge and experience gained from biplane design i... faced with the difficulty to obtain satisfactory longitudinal stability...

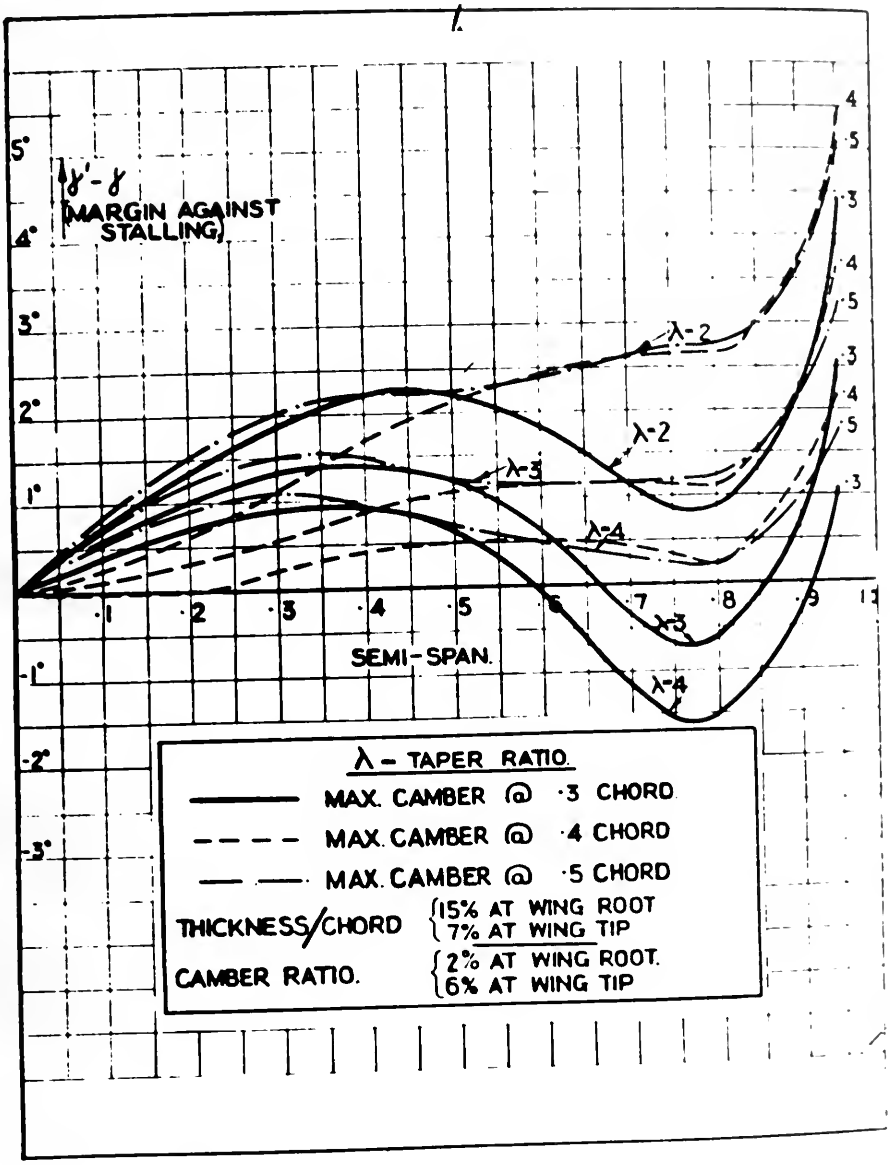


Fig. 13.

found it necessary either to shift the C.G. much more forward of the assumed position or to increase the tail volume considerably beyond which gave satisfactory stability on biplanes. There are various rea... account for this mysterious instability of the monoplane, and in th... I propose to deal with some of the major causes, but I am not claimi... effects mentioned are the only ones. The conclusions drawn are b... careful analysis of wind tunnel tests with a twin-engined monopla... taper ratio of about 4 : 1 and a tail volume of 0.55.

Fig. 14 shows a typical pitching moment diagram for a twin-en... plane. The resulting pitching moment has been resolved into mom...

Admitted November 24, 1950.

DEFENDANTS' EXHIBIT XX

District Court of the United States, Southern District of California, Central Division

Civil Action No. 10930-Y

MAURICE A. GARBELL, INC., a California Corporation, and GARBELL RESEARCH FOUNDATION, a California Corporation,

Plaintiffs,

vs.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION, a Delaware Corporation, and AMERICAN AIR LINES, INC., a Delaware Corporation,

Defendants.

STIPULATION #4

It is hereby stipulated subject to proof of error that the appended "Exhibit 22" is a reproduction of pages 604 to 613 of a printed publication "Luftfahrforshung" containing an article entitled "Elliptische Autriebsverteiling durch Verwindung und Profilanderung" published and issued by Z.W.B. in Berlin, Germany, in the year 1937 and that "Exhibit 22a" is a translation of said article (subject to correction if any error is contained therein), and that said "Exhibit 22a" may be used in evidence with the same force and effect as an original, subject to any objection which may be made thereto as irrelevant or immaterial when offered in evidence, viz:

LYON & LYON,
/s/ FREDERICK W. LYON,
Attorneys for Plaintiffs.

/s/ ROBERT B. WATTS,
/s/ FRED GERLACH,
Attorneys for Defendants.

C + Ic — Exhibit 22a

(Translation from Luftfahrtforschung, ... 1-15, 19..)

ELLIPTICAL LIFT DISTRIBUTION BY TWIST AND CHANGE IN PROFIL

by Shih Cheng Zien, Shanghai[1]

(1. Thesis, Technical University, Berlin.)

Abstract

An examination is made of the methods by which elliptical lift distribution can be attained spanwise by twist and profile variation. By this means air flow separation (or burbling) occurs at the wing tips later than in the center of the wing.

Elliptical lift distribution gives the smallest induced drag [?].[2]

(2. The numbers given in brackets refer to "References", Section IX.)

Lateral stability is guaranteed even at stall, by the delayed separation of the flow at the wing tips whereby the danger of spin [?] is reduced.

The trapezoidal wing has a simple planform. Highly tapered trapezoidal wings have especially greater depth at the root. Thereby, the stiffness is increased (singularly favorable for wing vibration) and the weight is reduced. This construction permits the useful load being placed in the center of the wing.

Contents

I. Fundamentals of Airfoil Theory

II. Fuch's Solution

III. Approximation of the Wing Contour

IV. Calculation of Twist for an Elliptical Lift Distribution

1. Analytical Solution
2. Graphical Solution
3. Comparison of the Analytical and Graphical Solutions.
4. Discussion of the Results from Twist

V. Determination of the Angle of Attack to which the change in profile corresponds for an Elliptical Lift Distribution.

1. c_a' Spanwise Distribution and the Influence of c_a
2. Profile Systematics

EXHIBIT 22a C 21

. Graphical Methods for the Determination of the Distribution of the Angle of Attack
. Analytical Test of the Lift Distribution and the Increase in Induced Drag

omparison of a Non-Twisted Elliptical Airfoil and the Trapezoidal ing with Twist and Change in Profile.

. Induced Drag
. Flow Separation, Lateral Stability and Lift Loss
. Comparison with Experimental Results Formerly Obtained

wist for Any Lift Distribution

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ppendix

I. Fundamentals of Airfoil Theory

he Lift of a portion of a wing of infinite span, having the t dx, is given by Kutta-Joukowski's Circulation Theorem

$$dA = \rho v_\infty \Gamma(x)\, dx \tag{1}$$

e:

ρ = the air density

v_∞ = the stream velocity at infinity.

$\Gamma(x)$ = the circulation at the point x.

Practically, the lift is calculated by the formula:

$$dA = C_a(x) \frac{\rho}{2} v_\infty^2\, t(x)\, dx \tag{2}$$

e:

C_a = the value of the lift (determined experimentally) at the point x.

$t(x)$ = the wing chord at the point x.

A comparison of equations (1) and (2) gives:

$$\Gamma(x) = \frac{1}{2} C_a(x)\, t(x)\, v_\infty \tag{3}$$

The circulation Γ is proportional to the product of C_a and t. C_a roportional to the angle of attack, α, relative to the axis of lift.

$$c_a = \frac{dc_a}{d\alpha} \cdot \alpha$$

rculation distribution for a wing of infinite span is directly
tional to the angle of attack and the wing chord.

e lift distribution for airfoils of finite span is calculated by
l's Method [2]. The circulation is here no longer proportional
geometrical but to the effective angle of attack, α. The
ence between the geometrical angle of attack, α_g and the effective,
s the induced angle of attack, α_i

$$\alpha_i = \frac{v_i(x)}{v_\infty} = \frac{1}{4\pi v_\infty} \int_{-\frac{b}{2}}^{+\frac{b}{2}} \frac{d\Gamma}{d\xi} \cdot \frac{d\xi}{x - \xi} \qquad (5)$$

x is the point at which the induced angle of attack is calculated.
is the abscissa, variable over the span. The effective angle of
thus is then:

$$\alpha_e = \alpha_g - \alpha_i$$

ubstitution of α_e in equations (3) and (4) gives:

$$\Gamma(x) = \tfrac{1}{2} c'_{a\infty} (\alpha_g - \alpha_i)\, t(x) \cdot v_\infty \qquad (6)$$

ith reference to equation (5)

$$) = \tfrac{1}{2} t(x) v_\infty c'_{a\infty} \left[\alpha_g(x) - \frac{1}{4\pi v_\infty} \int_{-\frac{b}{2}}^{+\frac{b}{2}} \frac{d\Gamma}{d\xi} \cdot \frac{d\xi}{x - \xi} \right] \qquad (7)^3$$

See, Fuchs-Hopf-Seewald: Aerodynamics Vol. II, Chapter V, pp. 139-140)

irculation is determined spanwise by this integral equation when
ing contour and the distribution of the angle of attack are given.

II. Fuchs' Solution [1]

quation (7) was solved by Betz [4] by means of a power series, by
rt [6] and Lotz [5] by means of a Fourier Series, by Fuchs [1] by
of a trigonometrical polynomials and graphically by Lippisch [7].
n Fuchs' method the airfoil contour is approximated as well as
ble by the fewest possible members of a trigonometric polynomial
practical wing model, the approximate contour possesses thereby
h leading and trailing edges, as well as rounded wing tips. This
dvantageous compared to the zigzag sinusoidal wing edges for the

ximation of the contour by other methods.

quation (7) is simplified by the introduction of new variables:

$$x = -\frac{b}{2}\cos\varphi \quad , \quad \xi = -\frac{b}{2}\cos\psi$$

φ and ψ vary from 0 to π, when x and ξ vary from $(-\frac{b}{2})$ t

); furthermore $\Gamma(x) = 2 b v_\infty G(x)$.

$$t = \frac{4b}{c'_{a\infty}} \mu(x)$$

e:

$$G(\varphi) = \mu(\varphi)\left[\alpha_g(\varphi) - \frac{1}{\pi}\int_0^\pi \frac{dG}{d\psi}\frac{d\psi}{\cos\psi - \cos\varphi}\right] \quad (8)$$

ontour function $\mu(\varphi)$ is an odd sine function with odd members, if irfoil is symmetrical about the center, $\varphi = \frac{\pi}{2}$., and decreases ds the wing tips.

$$\mu(\varphi) = \mu_1 \sin\varphi + \mu_3 \sin 3\varphi + \mu_5 \sin 5\varphi + \dots\dots$$

arly for the circulation

$$G(\varphi) = G_1 \sin\varphi + G_3 \sin 3\varphi + G_5 \sin 5\varphi + \dots\dots$$

relation transforms (8) into:

$$\varphi\sum_{n=1}^{\infty} G_{2n-1} \sin(2n-1)\varphi = \alpha_g \mu \sin\varphi - \mu\sum_{n=1}^{\infty}(2n-1)G_{2n-1}\sin(2n-1)\varphi \quad (9)$$

he geometrical angle of attack, α_g, is represented in the al case by:

$$\alpha_g(\varphi) = \alpha_0 + \alpha_2 \cos 2\varphi + \alpha_4 \cos 4\varphi + \dots\dots \quad (10)$$

is symmetrical about the wing center and decreases towards the wing

he evaluation of the coefficients $G_1, G_3 \dots$ according to equation (9) is:

$$\left.\begin{aligned} G_1 &= S_1 - \sum_{\lambda=1}^{\infty}(2\lambda-1)\mu_{2\lambda-1}G_{2\lambda-1} \\ G_{2k+1} - G_{2k-1} &= S_{2k+1} - S_{2k+1} \,. \\ &\quad - \sum_{\lambda=1}^{\infty}(2\lambda-1)(\mu_{2\lambda+2k-1} \pm \mu_{2\lambda-2k-1})G_{2\lambda-1} \end{aligned}\right\} \quad (11)$$

the minus sign is valid as long as $\lambda \leqq k$ and plus if $\lambda > k$, so one takes: $\mu_{-1} = -\mu_1$, $\mu_{-3} = -\mu_3$.

his way:

$$S_{2i+1} = \alpha_0 \mu_{2i+1} + \frac{\alpha_2}{2}(\mu_{2i+3} + \mu_{2i-1}) + \frac{\alpha_4}{2}(\mu_{2i+5} + \mu_{2i-3}) + \dots\dots$$

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$: = 2$

$$\left.\begin{aligned} &\mu_1)G_1 + 3\mu_3 G_3 + 5\mu_5 G_5 = S_1 \\ &\mu_3 G_1 + [1 + 3(\mu_1 + \mu_3 + \mu_5)] G_3 + 5(\mu_3 + \mu_5) G_5 = S_3 \\ &\mu_5 G_1 + 3(\mu_3 + \mu_5) G_3 + [1 + 5(\mu_1 + \mu_3 + \mu_5) G_5 = S_5 \end{aligned}\right\} \qquad (12)$$

r:

$$S_1 = \alpha_0 \mu_1 + \frac{\alpha_2}{2}(\mu_3 - \mu_1) + \frac{\alpha_4}{2}(\mu_5 - \mu_3)$$

$$S_3 = \alpha_0 \mu_3 + \frac{\alpha_2}{2}(\mu_5 + \mu_1) + \frac{\alpha_4}{2}(-\mu_1)$$

$$S_5 = \alpha_0 \mu_5 + \frac{\alpha_2}{2}\mu_3 + \frac{\alpha_4}{2}\mu_1$$

he approximation of the contour gives us $\mu_1, \mu_3, \dots \mu_{k+1}$, the ximation of the twist $\alpha_0, \alpha_2 \dots \alpha_{2\lambda}$; we have therewith $(k+1)$ ions for the calculation of the $(k+1)$ unknown of the lift function $_3 \dots G_{2k+1}$. The series $\mu(\varphi), G(\varphi), \alpha_q(\varphi)$ are rapidly convergent [1]. he calculation of the lift, it is, in general sufficient to ximate three terms each for $\mu(\varphi)$ and $\alpha_q(\varphi)$ in order to solve for hree unknowns G_1, G_3, G_5 from the three linear equations. onversely, for a given lift distribution $G(\varphi)$ and a given wing ur $\mu(\varphi)$ the twist $\alpha_q(\varphi)$ can easily be calculated. Fuchs treats roblem: How must the airfoil be twisted for an elliptical lift ibution?

n this work Fuchs' proposal is further developed and, indeed that low separates at the tips later than in the center is considered. or the solution of the proposed problem, a series of assumed zoidal airfoils is investigated, in which the wing contour is ximated by several members of a trigonometric polynomial and the calculated thereby compared with the desired condition. n this work is given a method according to which all such ximations can easily be performed graphically.

III. Approximation of the Wing Contour

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The contour function:

$$\mu(\varphi) = \mu_1 \sin\varphi + \mu_3 \sin 3\varphi + \mu_5 \sin 5\varphi \quad \text{or}$$

$$t(\varphi) = t_1 \sin\varphi + t_3 \sin 3\varphi + t_5 \sin 5\varphi$$

i hereby wanted so that $t(\varphi)$ accurately defines the airfoil surface id represents as far as possible an experimental wing contour.

The first coefficient μ_1 or t_1 is given analytically by the condition ' the equality of the surfaces:

$$F = \int_{-\frac{b}{2}}^{+\frac{b}{2}} t(x)\,dx = \frac{b}{2}\int_0^{\pi} t(\varphi)\sin\varphi\, d\varphi \doteq \frac{b}{4}\cdot\pi\cdot t_1$$

$$t_1 = \frac{4F}{\pi b}$$

$$\mu_1 = \frac{c'_{a\infty}}{4b}\cdot t_1 = \frac{c'_{a\infty}}{\pi A} \qquad \left(A = \frac{b^2}{F}\right)$$

The members of higher order are without influence on the surface rea; they are a function only of the chord distribution. They are raphically determined.

The half span is obtained from the abscissa, the wing chord from ne ordinate (See Appendix, Fig. 1).

ne semi-span is subdivided in the cosine of the angle varying by nits of 10°. The cosine division is obtained quickly and accurately f a quarter-circle with radius $r = \frac{b}{2}$ is drawn below the figure, the uarter circle is divided into nine equal parts and from the parts btained in this way, perpendiculars are dropped on-to the base. It s recommended that the scale of the diagram be chosen so that $\frac{b}{2}$ s approximately 20 to 30 cm.

The cosine division of the abscissa is plotted twice on transparent aper. The ellipse

$$y_1 = t_1 \sin\varphi$$

s drawn over one of the cosine divisions. The function

$$y_3 = t_3 \sin 3\varphi$$

s superimposed in this ellipse for various t_3's. It is sufficient n most cases to put:

$$\frac{2t_3}{b} = \pm 0.05, \ \pm 0.10, \ \pm 0.15, \ \pm 0.20, \ \pm 0.25$$

e Appendix, Fig. 2)

The transparent paper is then laid on the figure on which the actual
rd is plotted and one judges which curve y or which t_3 best
rresponds to the actual airfoil contour. The first approximation of
t_3 is determined sufficiently accurate by interpolation of the
dividual curves:

$$\mu_3 = \frac{c'_{a\infty}}{4b} \cdot t_3$$

e curve

$$y = t_1 \sin\varphi + t_3 \sin 3\varphi$$

plotted on the other cosine division where t_3 corresponds to the
lue just found by interpolation. The function

$$y_5 = t_5 \sin 5\varphi$$

plotted over this curve for different t_5's. It is sufficient to
ot:

$$\frac{2t_5}{b} = \pm 0.025, \pm 0.050, \ \pm 0.075, \ \pm 0.100$$

e Appendix, Fig. 3)

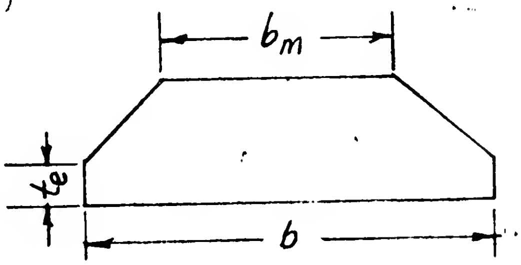

Fig. 1: On the Trapezoidal Ratio

This transparent paper is now laid over the figure on which the
ctual wing chord is plotted and one judges which curve y or t_3
orresponds best with the actual outline:

$$\mu_5 = \frac{c'_{a\infty}}{4b} \cdot t_5$$

The values of y_1, y_3, y_5 are obtained quickly and accurately in
rawing circles about a point with radii t_1, t_3, t_5 and the

diculars at every 10°. Firstly, it can be established for the ination of t_5 if this first approximation for t_3 was well chosen. , the process must be repeated, i.e. t_3 and t_5 are again ined. As the actual wing contour can be scrutinized each time if the first approximation is as good as the former. It is sufficient practically if only the first three members of the ometric series are used.

n the present work, 26 trapezoidal airfoils with the same area, the ratio of sides ($\Lambda = 5$) but different trapezoidal ratios were tigated. (See Appendix, Table 1 examples for that purpose, dix, Figs. 4 to 8)

rapezoidal ratio, Fig. 1.

$$\frac{b_m}{b} = 0;\ 0.2;\ 0.4;\ 0.6;\ 0.8;\ 1$$

$$\frac{t_e}{t_m} = 0;\ 0.2;\ 0.4;\ 0.6;\ 0.8$$

he dimensionless coefficients, μ_1, μ_3, μ_5 of the contour function nversely proportional to the ratio of the sides Λ. For other os of the sides, μ_1, μ_3 and μ_5 must change correspondingly.

IV. Calculation of Twist for an Elliptical Lift Distribution

The twist function

$$\alpha_q(\varphi) = \alpha_0 + \alpha_2 \cos 2\varphi + \alpha_4 \cos 4\varphi$$

o be found.

The contour functions

$$\mu(\varphi) = \mu_1 \sin \varphi + \mu_3 \sin 3\varphi + \mu_5 \sin 5\varphi$$

the condition, that the circulation distribution shall be elliptical,

$$G(\varphi) = G_1 \sin \varphi$$

$C'_{a\infty}$ = constant spanwise

given.

Analytical Solution

If, in equation (12),

$$G_3 = G_5 = 0$$

bstituted, then

$$\left.\begin{aligned} \mu_1) G_1 &= \alpha_0 \mu_1 + \frac{\alpha_2}{2}(\mu_3 - \mu_1) + \frac{\alpha_4}{2}(\mu_5 - \mu_3) \\ \mu_3 G_1 &= \alpha_0 \mu_3 + \frac{\alpha_2}{2}(\mu_5 + \mu_1) + \frac{\alpha_4}{2}(-\mu_1) \\ \mu_5 G_1 &= \alpha_0 \mu_5 + \frac{\alpha_2}{2}\mu_3 \qquad + \frac{\alpha_4}{2}\mu_1 \end{aligned}\right\} \qquad (13)$$

tained. The solution of the equations gives:

$$\left.\begin{aligned} \alpha_0 &= \left(1 + \frac{1}{\mu_1}\,\frac{q}{ps - rq}\right) G_1 \\ \alpha_2 &= \frac{2}{\mu_1}\cdot\frac{p}{ps - rq} G_1 \\ \alpha_4 &= \frac{2}{\mu_1}\,\frac{1}{ps - rq}\left[\frac{\mu_5}{\mu_1} q - \frac{\mu_3}{\mu_1} p\right] G_1 \end{aligned}\right\} \qquad (14)$$

$$p = \frac{\mu_3}{\mu_1} + \frac{\mu_5}{\mu_1} \qquad\qquad r = \frac{\mu_3}{\mu_1}\left(\frac{\mu_5}{\mu_1} - \frac{\mu_3}{\mu_1}\right) + 1$$

$$q = 1 + \frac{\mu_3}{\mu_1} + \frac{\mu_5}{\mu_1} \qquad\qquad s = \frac{\mu_5}{\mu_1}\left(\frac{\mu_5}{\mu_1} - \frac{\mu_3}{\mu_1} + 1\right) - 1$$

the numerical values for μ_1, μ_3, μ_5 are introduced into these ons, it is shown that q is much larger than p, p is much larger than $\left[\frac{\mu_5}{\mu_1} q - \frac{\mu_3}{\mu_1} p\right]$ ries $\alpha_q(\varphi)$ converges very rapidly, so that $\alpha_q(\varphi)$ is determined iently accurately by three terms. The twist sought is then:

$$= \alpha_0 + \alpha_2 \cos 2\varphi + \alpha_4 \cos 4\varphi$$

$$G_1\left[1 + \frac{1}{\mu_1}\,\frac{q}{ps - rq} + \frac{2}{\mu_1}\,\frac{p}{ps - rq}\cos 2\varphi + \frac{2}{\mu_1}\,\frac{1}{ps - rq}\left\{\frac{\mu_3}{\mu_1} q - \frac{\mu_5}{\mu_1} p\right\}\cos 4\varphi\right]$$

$$= G_1 + \frac{G_1}{\mu_1}\left[\frac{q}{ps - rq} + \frac{2p}{ps - rq}\cos 2\varphi + \frac{2}{ps - rq}\left\{\frac{\mu_5}{\mu_1} q - \frac{\mu_3}{\mu_1} p\right\}\cos 4\varphi\right]$$

e geometrical angle of attack is composed of two parts, the d angle of attack

$$\alpha_i = \frac{C_a}{\pi\Lambda} = G_1,$$

is constant spanwise, and the effective angle of attack which is le spanwise but which is everywhere proportional to

$$\frac{G_1}{\mu_1} = \frac{C_a/\pi\Lambda}{C'_{a\infty}/\pi\Lambda} = \frac{C_a}{C'_{a\infty}} = \alpha_e \text{ ellipt. Fl.}$$

is; moreover, the constant lift coefficient, which corresponds with lliptical airfoil contour, $\alpha_{e \text{ ellipt. Fl.}}$ is the accompanying constant effective of attack, then

$$(\varphi) = \alpha_i + \alpha_{e \text{ ellipt.Fl.}} \left[\frac{q}{ps-rq} + \frac{2p}{ps-rq} \cos 2\varphi + \frac{2}{ps-rq} \left\{ \frac{\mu_5}{\mu_1} q - \frac{\mu_3}{\mu_1} p \right\} \cos 4\varphi \right. \tag{15}$$

he twist function is calculated for the airfoils examined for $= 0.3 : C'_a = 2\pi\, 0.833$. The numerical values of the calculations are in Table 1 of the appendix; for that purpose, Fig. 4 to 8 of the dix are drawn as examples.

2. Graphical Solution

quation (9) is transformed into:

$$\alpha_q(\varphi) = \frac{G_1 \sin\varphi + G_3 \sin 3\varphi + G_5 \sin 5\varphi + \ldots\ldots}{\mu_1 \sin\varphi + \mu_3 \sin 3\varphi + \mu_5 \sin 5\varphi + \ldots\ldots} + \frac{G_1 \sin\varphi + 3G_3 \sin 3\varphi + 5G_5 \sin 5\varphi + \ldots\ldots}{\sin\varphi} \tag{16}$$

an elliptical distribution: $G_3 = G_5 = \ldots\ldots = 0$

$$\alpha_q(\varphi) = G_1 + \frac{G_1 \sin\varphi}{\mu_1 \sin\varphi + \mu_3 \sin 3\varphi + \mu_5 \sin 5\varphi}$$

elliptical wings, the effective angle of attack is

$$\alpha_{e \text{ ellipt.Fl.}} = \frac{G_1 \sin\varphi}{\mu_1 \sin\varphi} = \frac{G_1}{\mu_1}.$$

$$G_1 \sin\varphi = \alpha_{e \text{ ellipt.Fl.}} \cdot \mu_1 \sin\varphi$$

ce:

$$\alpha_q'(\varphi) = \alpha_i + \alpha_{e \text{ ellipt.Fl.}} \frac{\mu_1 \sin\varphi}{\mu_1 \sin\varphi + \mu_3 \sin 3\varphi + \mu_5 \sin 5\varphi} \tag{16a}$$

e

$\alpha_{e \text{ ellipt.Fl.}} = \dfrac{C_a}{C'_{a\infty}}$ = the constant effective angle of attack for elliptical wings.

$\alpha_i = \dfrac{C_a}{\pi\Lambda} = G_1$ = the constant induced angle of attack.

The distribution function of the effective angle of attack is obtained ivision of the wing chord of the elliptical airfoil by the roximated wing contour (See, appendix, as example, Fig. 4).

3. Comparison of the Analytical and Graphical Methods

Comparison of equations (15) and (16a) must yield agreement

oth distribution functions:

$$\frac{\;}{q}+\frac{2p}{ps-rq}\cos 2\varphi+\frac{2\left(\frac{\mu_5}{\mu_1}q-\frac{\mu_3}{\mu_1}p\right)}{ps-rq}\cos 4\varphi=\frac{\mu_1\sin\varphi}{\mu_1\sin\varphi+\mu_3\sin 3\varphi+\mu_5\sin 5\varphi}$$

For a special case, namely, the elliptical airfoil, i.e. $\mu_3 = \mu_5 = 0$

s the equation to be correct: both sides are unity. The curves

wist from the graphical process are somewhat smaller in the center,

gradually become larger towards the wing tips than those from the

ytical procedure. The greatest deviation between the analytical

graphical methods amounts to approximately 2% for rectangular wings

16% for delta wings. It can, therefore, be concluded that the

phical method is applicable only for rectangular and ellipsoidal

foils.

4. Discussion of the Twist Results.

For elliptically contoured wings, the angle of attack is

stant spanwise. For trapezoidal airfoils, with a taper ratio, $\frac{t_e}{t_m} = \frac{1}{3}$

e Appendix, Fig. 9), the angle of attack is the same in the center

at the wing tips.

$$\alpha_m = \alpha_{end}$$

For all trapezoidal airfoils with a taper ratio $\frac{t_e}{t_m} < \frac{1}{3}$, the

le of attack increases towards the tips. They are useless. The

ference between the angle of attack at the center of the wing and that

the wing tips attains its greatest value for delta wings $\Delta\alpha_g = 5.4°$

For all trapezoidal airfoils with a taper ratio $\frac{t_e}{t_m} > \frac{1}{3}$, the

le of attack decreases towards the tips. They are useful. The

ference between the angle of attack at the center and that at the

s becomes a maximum for rectangular airfoils, $\Delta\alpha_q = 2.2°$

Comparison with an elliptical wing gives a good appraisal of the

nwise distribution of twist. Wherever the chord of a trapez...

s greater than that of an elliptical (wing), the angle of attack
ller and conversely.

e mathematical condition therefor, that the angle of attack decreases
ically towards the wing tips is:

$$\frac{d\alpha_g}{dx} < 0$$

n our case,

$$\alpha_2 > 0 \quad or \quad p > 0$$

I the aspect ratio Λ increases, the geometrical angle of attack, $\alpha_g =$
$+\alpha_e$, decreases for the induced angle of attack, $\alpha_i = \frac{C_a}{\pi\Lambda}$, distributed
only spanwise, is inversely proportional to Λ , and the effective
e of attack

$$\alpha_e = \frac{C_a}{C'_{a\infty}} f(\mu, \varphi)$$

ependent of Λ . The taper ratio $\frac{t_e}{t_m} = \frac{1}{3}$ at $\alpha_m = \alpha_{end}$, is thus
for all trapezoidal wings haveing equal wing area and different
s

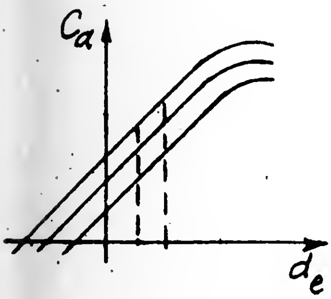

Fig. 2
the Influence of C_a

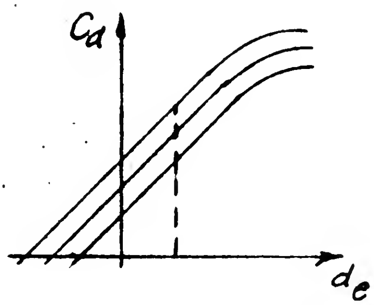

Fig. 3
On the Influence of C_a

I. Determination of the Angle of Attack which Corresponds to the Change in Profile for a Elliptical Lift Distribution

1. C_a Spanwise Distribution and the Influence of C_a.

From the condition that the lift $C_a \cdot t$ shall be elliptical
ise, a definite course of C'_a is given for each distribution of
hickness, t

$$C_a = C'_{a\infty} \alpha_e = C'_{a\infty} (\alpha + \beta)$$

β is the angle of zero lift C_a' is practically equal and constant
all profiles.

may vary in three ways.

a. One and the same profile is retained over the whole span and the angle of attack is varied so that a definite angle of attack belongs to one value of C_a and conversely

$$\alpha_e = \alpha_{e\ ellipt.Fl.} \frac{G_1 \sin \varphi}{\mu_1 \sin \varphi + \mu_3 \sin 3\varphi + \mu_5 \sin 5\varphi}$$

This problem corresponds to the twist in Chapter V. For a given distribution of C_a, the distribution of the angle of attack is determined uniquely.

b. The same angle of attack is retained along the entire span and the profile varied, Fig. 2, so that a different value of C_a corresponds to the same angle of attack; i.e. profiles with different zero lift angles β are available in practice. Thereby, a spanwise distribution of the angle of attack is arbitrarily given, and the profile sought, in order to obtain a definite distribution of C_a.

c. The angle of attack and profile are both varied (Figure 3). Thereby, a distribution of C_a is given and the profile and angle of attack are to be found. The latter two belong to changes in profile [9].

2. Profile Systematics

A profile [11, 12] is characterized by the magnitude and ition of the camber and the thickness ratio $\frac{\delta}{t}$. The greater the ber $\frac{f}{t}$, the greater becomes the zero lift angle β and the maximum t coefficient, $C_{a\,max}$. The farther the maximum camber line lies rwards, the farther to the rear is the center of pressure.

The greater the thickness ratio, $\frac{\delta}{t}$, the smaller is $\frac{dc_a}{d\alpha}$. The maximum value of the lift coefficient $C_{a\,max}$ increases at first with the thickness ratio, reaches a maximum at approximately $\frac{\delta}{t} = 0.12$ and then decreases again, where (Fig. 4)

β the angle of zero lift; α the angle of attack

α_1 the angle of attack referred to the axis of zero lift.

3. Graphical Method for the Evaluation of the Distribution of the Angle of Attack

Given the spanwise distribution of C_a and the condition that the angle of attack as well as the thickness ratio must decrease towards the wing tips.

The profiles and the geometrical angle of attack at each position are to be found.

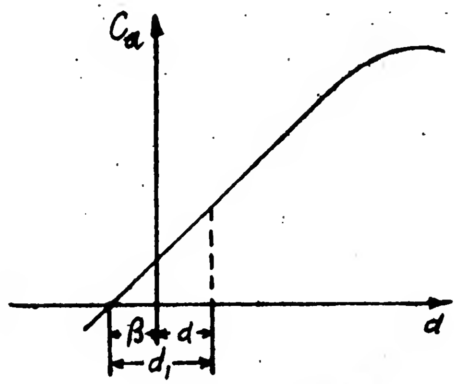

Fig. 4: On Profile Systematics

The greater the zero lift angle of a profile, the smaller is the angle of attack which is due to a definite value of C_a and the less is the danger of flow separation. It is recommendable that the angle of zero lift of the profile be as large as possible at the wing tips, i.e. in practice the camber of the profile shall be proportionally large at the wing tips.

It is sufficient here if the profiles and angles of attack are evaluated at five stations of the semi-wing. The value of C_a is calculated, the profile chosen, the effective angle of attack read off

$=f(\alpha)$ curve, the induced angle of attack, $\alpha_i = \frac{C_a}{\pi\Lambda}$ calculated and the geometrical angle of attack, $\alpha_g = \alpha_i + \alpha_e$, obtained. (See 10 to 12, Numerical Table 2).

or the determination of the range of the angle of attack up to flow ion for different locations of the wing, the profiles are plotted on n geometrical angle of attack by a point in Figure 12 Appendix, on the profiles are adjusted for fast flight. Up to $C_{a\,max}$ the NACA 0021 has a much smaller range of the angle of attack in the center of the han NACA 6409 profile has at the wing tips. In consequence the flow tes first in the center of the wing, then gradually outwards to the tips.

special case is where the profiles are to be found for a given bution of the angle of attack, e.g. for a distribution of the angle of decreasing linearly from the wing center to the tips. For a value and for an effective angle of attack, a definite point in the $c_a = f(\alpha)$ can be measured. By interpolation, the profile can be determined, Fig. 5. y this method, an elliptical lift distribution can be attained for apezoidal wings, for the difference in the angle of attack between ng center and the tips can be made.

Example

x =	0 (Wing Center)	$\frac{b}{8}$	$\frac{b}{4}$	$\frac{3b}{8}$	$\frac{b}{2}$ (Wing Tip)	Comparison Profile
ted NACA profile	0021	2418	4415	6412	6409	0015
r f/t in percent	0	2	4	6	6	0
of zero lift, β°	-0.1	-1.9	-3.8	-5.7	-5.9	0
ness Ratio, δ/t in %	21	18	15	12	09	15
$\frac{c_a}{\alpha}$, per degree	0.094	0.098	0.100	0.101	0.101	0.100
of maximum lift, $\alpha_{C_{a\,max}}$	17	15	15	15	15	17
ated effective angle attack, α_e degrees	2.92	0.92	-0.55	-2.19	-2.9	
lated Induced Angle Attack, α_i degrees	0.913	0.913	0.913	0.913	0.913	
trical Angle of ack, α_g degrees	3.833	1.833	0.363	-1.277	-1.987	
of Angle of Attack m the Angle of zero t to the Angle of $C_{a\,max}$						
$^\circ = \alpha^\circ_{C_{a\,max}} - \alpha^\circ_{C_a=0}$, degrees	17.1	16.9	18.8	20.7	20.9	

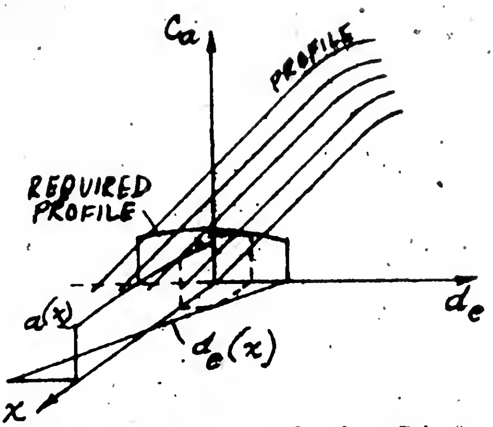

Fig. 5: On the determination of the Distribution of the Angle of Attack as great as the cambers of the profile permit.

4. Mathematical Examination of the Lift Distribution and of the Increase of Induced Drag

Two omissions are made in this method. The distribution of C_a is calculated from the approximate contour function. The coefficients of the contour function are determined by assuming that the mean value of the selected profiles $\frac{dc_a}{d\alpha}$ is constant spanwise. In reality, $\frac{dc_a}{d\alpha}$ vary somewhat for different profiles.

For the evaluation of the lift coefficient and of the angle of attack the flow around the wing has been considered as a plane problem. Actually the individual cross-sections mutually influence one another (space problem).

Whether these omissions are permissible will be verified by calculating the deviation of the lift distribution from the elliptical and the increase in the induced drag, which results from a profile with a mean $\frac{dc_a}{d\alpha}$ under the condition that the lift coefficient be invariable for the calculated five stations of the wing.

Example: Trapezoidal Wing: $\frac{t_e}{t_m} = 0.25$ $G_1 = 0.913$ $G_3 = -0.01404$

$G_3 = 0.00253$ $\frac{G_3}{G_1} = -0.01541$ $\frac{G_5}{G_1} = 0.00278$

$$\frac{\Delta c_{wi}}{c_{wi\,min}} = 3\left(\frac{G_3}{G_1}\right)^2 + 5\left(\frac{G_5}{G_1}\right)^2 = 0.00075 < 0.1\%$$

The increase in the induced drag compared to the smallest induced drag is very small.

VI. Comparison of an Untwisted Elliptical Wing and a Trapezoidal Wing with Twist and Profile Variation.

1. Induced Drag.

For elliptically contoured wings, the lift distribution lliptical under all flight conditions. For wings with any contour which an elliptical distribution is attained by twist or profile ation, the lift distribution is elliptical under one flight condition , generally under rapid flight. In the first case, the increase ift is proportional to t, in the second case, proportional to $t \cdot \frac{dc_a}{d\alpha}$. over, the Ca_{max} values (19) and the corresponding increase of the e of attack are ascertained. The lift distribution and the increase nduced drag are then calculated. It became evident that the ease in the induced drag is less than one percent (1%), in general, for Ca_{max}, on account of this deviation of the lift distribution the elliptical. The smaller the trapezoid ratio, the more vorable are the relations.

2. Flow Separation, Lateral Stability and Distribution Loss

For an elliptical wing with constant angle of attack wise, the flow separates almost simultaneously at all points, i.e. profiles attain their maximum lift values simultaneously. Lateral bility at stall is poor.

A trapezoidal wing with an angle of attack decreasing from the wing ter to the tips has only one place in the center where the flow rates first and a Ca_{max} value appears. Not every profile attains maximum lift coefficient, because the total lift is smaller for a pezoidal wing than for an elliptical wing with the same wing area.

3. Comparison with Former Experimental Results.

C. B. Millikan [12] has experimentally established that w separation occurs for a rectangular wing first in the center of the g, for a trapezoidal wing first at the tips, moreover invariably the rear edge. Prandtl [16] previously found the same results.

the case of elliptical wings the flow has to separate everywhere ultaneously, but Prandtl showed in his experiments that the flow arates first at the tips. Irving [13] tested trapezoidal wings h straight leading and trailing edges. It was shown that the flow arates in the case of a trapezoidal wing, in the center of the g having a straight leading edge and in the rear third of the wing ing a straight trailing edge. Huebner [17] calculated theoretically t the loss in lift for a conventional trapezoidal wing with a constant file is approximately one percent (1%) compared to an elliptical wing equal area. A. E. Lombard [14] established, in wind tunnel tests t the flow for a strongly tapered trapezoidal wing, having a tip profile large $C_{a_{max}}$ value, separates first in the center. Even in flight, observed that the stability is satisfactory. I. H Crowe [15] firmed that a twist of 8° is sufficient to prevent premature separation the flow at the wing tips. Large values improve the stability but reases the profile drag too much.

From the experimental results it appears:

1. That the flow separation at the wing tips is limited not only by the attitudes of an individual cross-section but also by the wing form and the lift distribution altered simultaneously thereby (See Appendix, Table 3).
2. That for the prevention of premature flow separation at the wing tips of rectangular wings, no twist is needed compared to the greater twist of about 6° to 8° for strongly tapered trapezoidal wings.

According to the results of calculation for an elliptical lift tribution in Chapters IV and V, the twist for an approximately tangular wing amounts to about 2° (See Appendix, Figure 9 $\frac{t_e}{t_m} = 1$), a strongly tapered trapezoidal wing to about 6° (See Appendix, Figure 1.) The twist is according to the results of calculation, somewhat ater for a rectangular wing, somewhat smaller for a strongly tapered

ezoidal wing than that of experimental results hitherto obtained. The twisted rectangular wing is thus more advantageous than the isted and has therefore more lateral stability. The flow separation strongly tapered trapezoidal wings with a change in profile begins t in the center and proceeds then gradually outwards to the tips Chapter V, 3). It is influenced by the trapezoidal form, according hether it is provided with straight leading or trailing edges, as ng [13] established experimentally.

The results of calculation for an elliptical lift distribution and experimental results previously reported agree sufficiently well in e of small deviations.

The development of the JU-86 wing, reported by A. W. Quick [19] in 1936 Yearbook of the Lilienthal Company for Aeronautical Research n good agreement with my experimental results.

VII. Twist for an Arbitrary Lift Distribution

If the lift distribution is known and non-elliptical, the ficients G_1, G_3, and G_5 of the circulation distribution can ys be determined by the same graphical method as the coefficients he contour function, t_1, t_3, and t_5. An arbitrary lift ribution has been resolved into one elliptical, $G_1 \sin\varphi$ and two lift distributions, $G_3 \sin 3\varphi$ and $G_5 \sin 5\varphi$ while the integrals:

$$\int_{-\frac{b}{2}}^{+\frac{b}{2}} G_3 \sin 3\varphi \, dx = 0 \quad ; \quad \int_{-\frac{b}{2}}^{+\frac{b}{2}} G_5 \sin 5\varphi \, dx = 0$$

The values of G_1, G_3, and G_5 are substituted in equation (12) thereby three equations in three unknowns α_0, α_2, and α_4 are n obtained.

VIII. Summary

In the case of a trapezoidal wing with a taper ratio, $\frac{t_e}{t_m} > \frac{1}{3}$, a elliptical distribution with good lateral stability at stall an be obtained by twist or profile change or both.

For a trapezoidal wing with a taper ratio, $\frac{t_e}{t_m} < \frac{1}{3}$ an elliptical stribution without endangering the lateral stability can be attained ly by variation in profile or by change in profile and twist. The teral stability is somewhat better for a weakly tapered trapezoidal ng than for a strongly tapered one.

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X Appendix

Fig. 2 On the approximation of the Contour

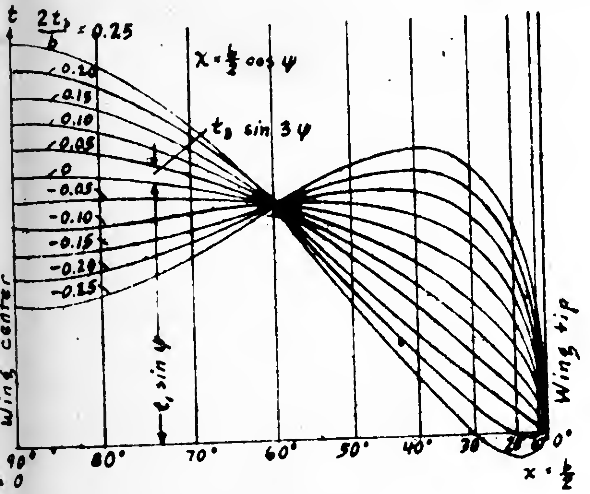
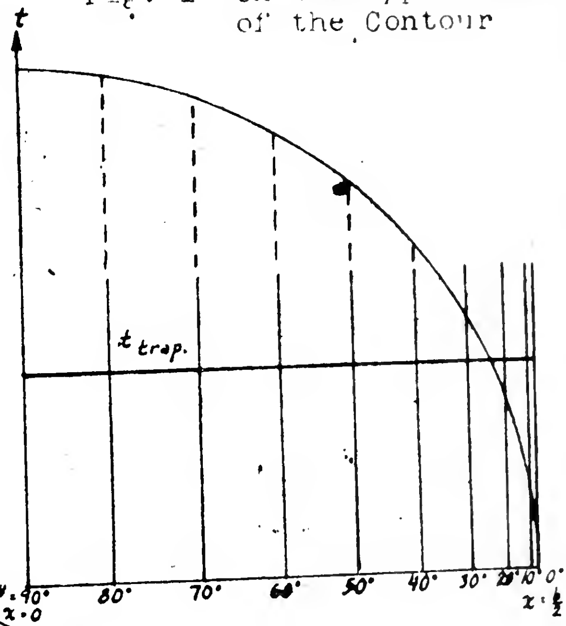


Fig. 1 On the approximation of the Contour



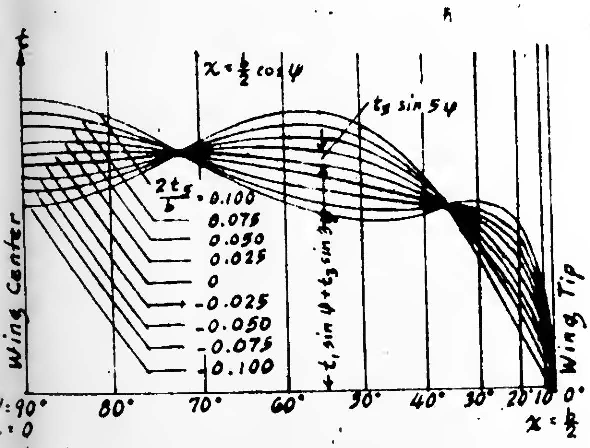


Fig. 3 On the approximation of the Contour

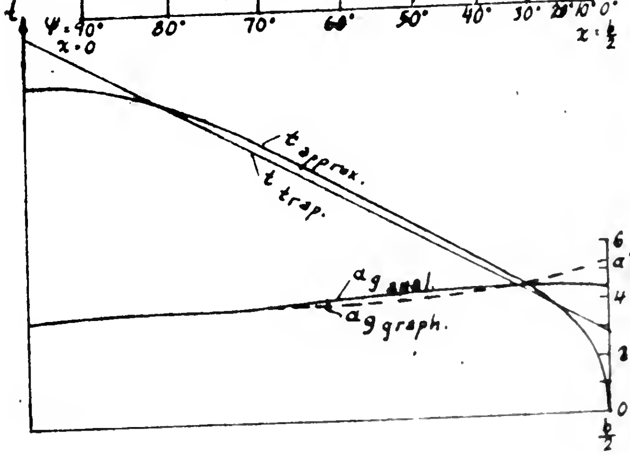


Fig. 4 Calculation of Twist Trapezoidal Ratio $b_m/b = 0$, $t_e/t_m = 0.2$, $\lambda = 5$

Calculation of Twist: Trapezoid Ratio

$\frac{b_m}{b} = 0.2, \frac{t_e}{t_m} = 0.4, \lambda = 5$

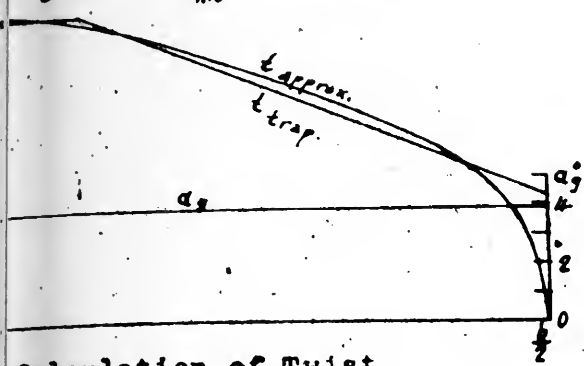
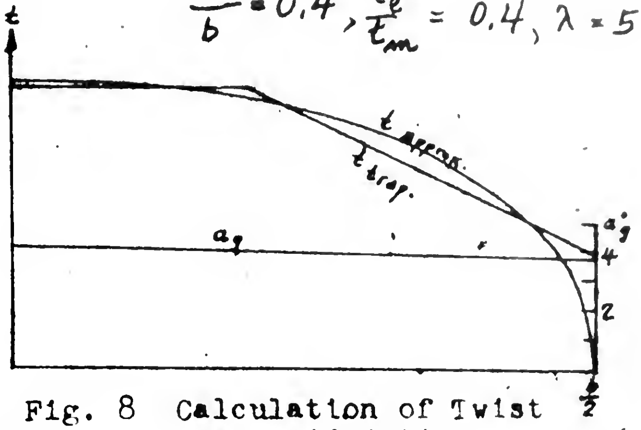


Fig. 6 Calculation of Twist Trapezoid Ratio

$\frac{b_m}{b} = 0.4, \frac{t_e}{t_m} = 0.4, \lambda = 5$



Calculation of Twist Trapezoid Ratio

$\frac{b_m}{b} = 0.6, \frac{t_e}{t_m} = 0, \lambda = 5$

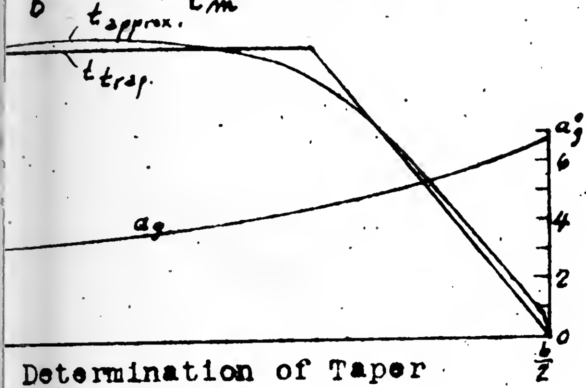
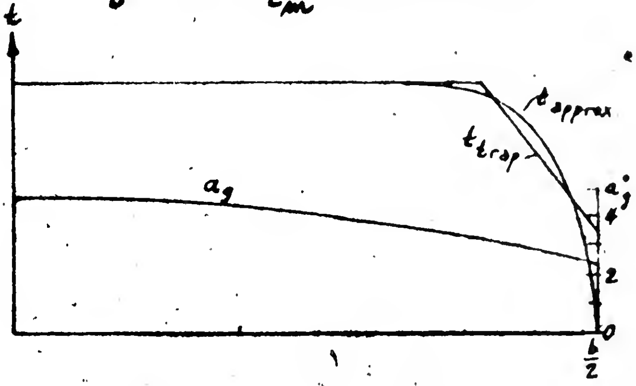


Fig. 8 Calculation of Twist Trapezoid Ratio

$\frac{b_m}{b} = 0.8, \frac{t_e}{t_m} = 0.4, \lambda = 5$



Determination of Taper Ratio For which

$a_m = a_{end}, \frac{t_a}{t_m} \approx \frac{1}{3}, \left(\frac{b_m}{b} = 0\right)$

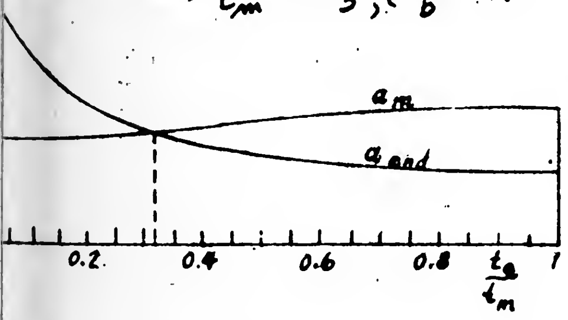


Fig. 10 Distribution of Angle of Attack by Change in Profile

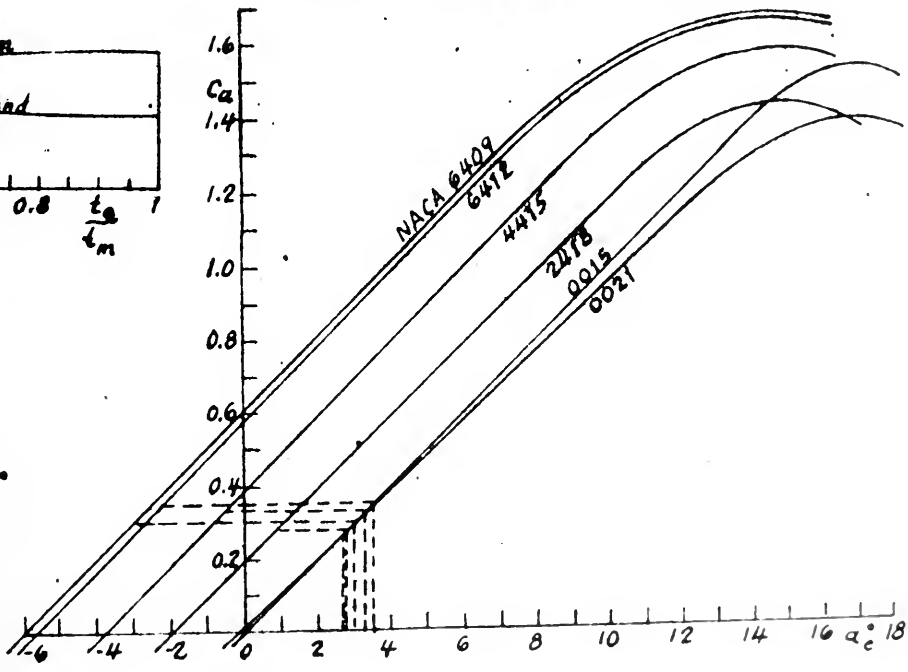


Fig. 11 Distribution of Angle of Attack by Change in Profile Trapezoid Ratio

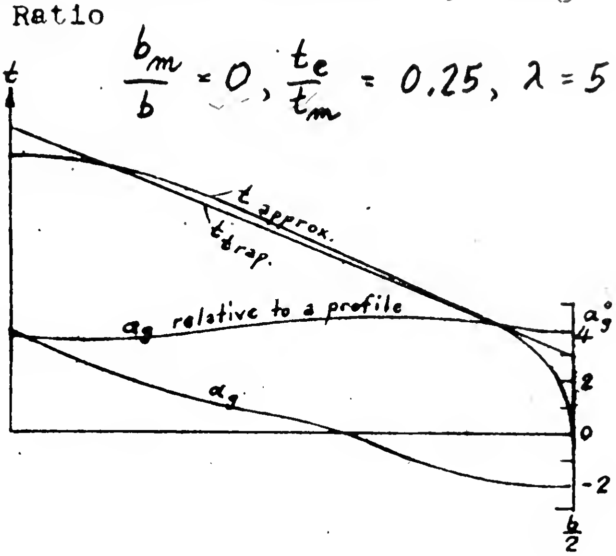
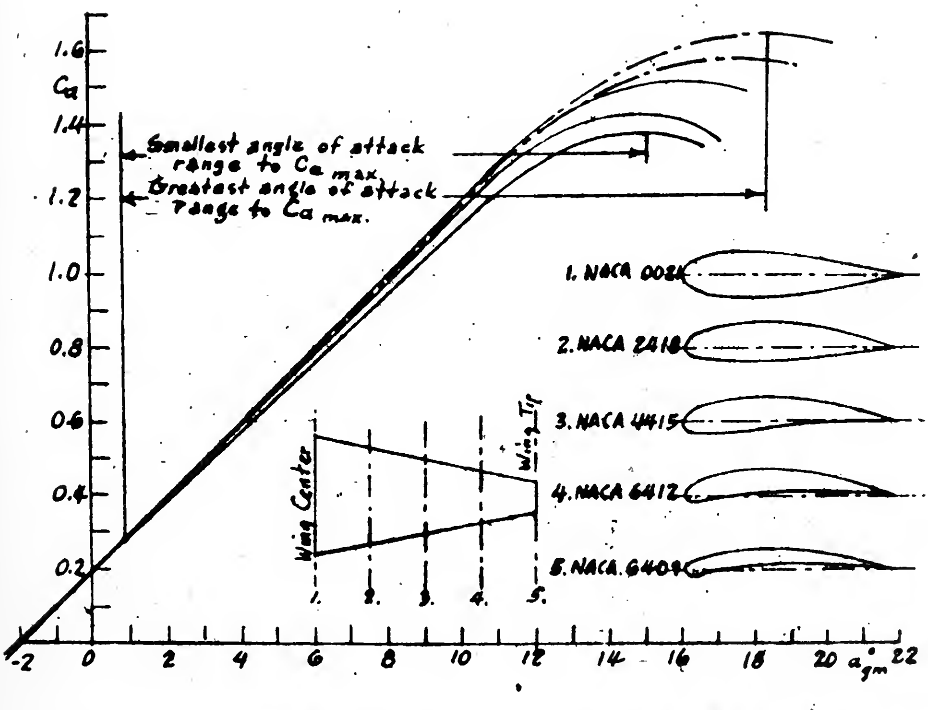


Fig. 12 Range of Angle of Attack to Flow Separation for Different Profiles



Mean Geometrical Angle of Attack

TABLE I

[C]alculation of the Twist ($\Lambda = 5$; $c'_\alpha = 2\pi\, 0.833$).

[..]d	t_1	t_3	t_5	μ_1	$\frac{\mu_3}{\mu_1}$	$\frac{\mu_5}{\mu_1}$	a_0	a_2	a_4
	cm	cm	cm				Deg.	Deg.	Deg.
	10.185	-1.308	0	0.333	-0.222	0	5.707	2.682	0.592
	10.185	-1.00	0.45	0.333	-0.0971	0.044	4.200	0.340	-0.234
	10.185	-0.33	0.40	0.333	-0.0324	0.039	3.950	-0.041	-0.234
	10.185	1.54	0.45	0.333	0.1512	0.0471	3.640	-0.863	-0.104
	10.185	1.80	0.50	0.333	0.177	0.0491	3.620	-0.967	-0.084
	10.185	2.12	0.50	0.333	0.208	0.0491	3.580	-1.055	-0.033
	10.185	-2.55	0	0.333	-0.25	0	5.960	3.310	0.795
	10.185	-1.45	0.50	0.333	-0.142	0.0491	4.410	0.750	-0.247
	10.185	0.50	0	0.333	0.0491	0	3.870	-0.282	-0.013
	10.185	1.00	0.50	0.333	0.0981	0.0491	3.680	-0.679	-0.067
	10.185	1.45	0.45	0.333	0.1424	0.0442	3.640	-0.830	-0.196
	10.185	-2.00	-0.50	0.333	-0.1965	-0.0491	[illegible].710	3.050	1.061
	10.185	-0.80	0.25	0.333	0.0786	0.0245	4.210	0.365	-0.127
	10.185	0.31	0	0.333	0.0304	0	3.930	0.176	0.052
	10.185	1.00	0.20	0.333	0.0982	0.0196	3.739	-0.576	-0.050
	10.185	1.70	0.28	0.333	0.1670	0.0275	3.640	-0.858	-0.002
	10.185	-0.85	-1.00	0.333	0.0835	0.0982	4.941	1.752	0.121
	10.185	-0.35	0.30	0.333	-0.0344	0.0293	4.020	0.051	-0.176
	10.185	0.75	-0.25	0.333	0.0736	0.0245	3.760	-0.495	-0.099
	10.185	1.75	0.25	0.333	0.1720	0.0245	3.650	-0.772	0.019
	10.185	2.00	0.30	0.333	0.1965	0.0295	3.596	-1.012	0.042
	10.185	1.30	0	0.333	0.1275	0	3.740	-0.620	0.078
	10.185	1.75	0.25	0.333	0.1720	0.0245	3.650	0.870	0.019
	10.185	2.10	0.40	0.333	0.2061	0.0393	3.595	-1.021	0.016
	10.185	2.40	0.60	0.333	0.2360	0.0589	3.550	-1.162	-0.030
	10.185	2.15	0.50	0.333	0.2101	0.0491	3.580	-1.001	-0.026

TABLE II

[C]hange in Profile: $\Lambda = 6$; $C'_{a\infty} = 2\pi\, 0.91$

	t_1 cm	t_3 cm	t_5 cm	μ_1	μ_3	μ_5	0 Deg.	2 Deg.	4 Deg.
5	9.3	-0.8	0.6	0.303	-0.026	0.0195	3.95	0	-0.37

TABLE III

...ison of Traction Surfaces With and Without Twist

	Wing with Twist	Wing Without Twist
...bution	Elliptical only for a definite value of C_a	Elliptical only for wings with Elliptical Contour. For all other Forms of Wings, the Lift Distribution is not Elliptical. Rectangular Wing (Appendix Fig. 13) Trapezoidal Wing (Appendix Fig. 14) (strongly tapered)
...ion of ...ration	Always in the wing center, because the range of the angle of attack to $C_{a\,max.}$ is less in the center than at the tips	Contour for Rectangular Wings (Appendix Fig. 15) — Position of Separation in the Center For elliptical wings (Appendix Fig. 16) at the wing tips. For Trapezoidal wings with straight leading edges (Appendix Fig. 17) — In the Center (rear) For Trapezoidal wings with straight trailing edges. (Appendix Fig. 18 — In the rear third of the outer wings.
...ange ...$_{max.}$	Not Simultaneous first in the center Small mean C_a	Theoretically, $C_{a\,max.}$ must be everywhere simultaneous. Experiments not simultaneous. Rectangular wings in the center. Trapezoidal wings and Elliptical Wings at the Tips. Larger mean C_a

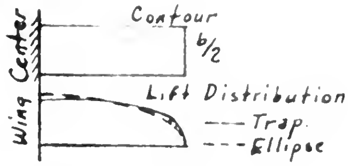


Fig. 13

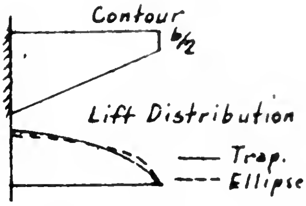


Fig. 14

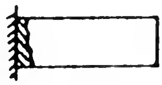


Fig. 15


Fig. 16

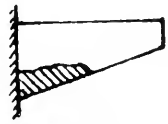


Fig. 17

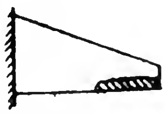


Fig. 18

Admitted November 24, 1950.

DEFENDANTS' EXHIBIT AAA

District Court of the United States, Southern District of California, Central Division

Civil Action No. 10930-Y

MAURICE A. GARBELL, INC., a California Corporation, and GARBELL RESEARCH FOUNDATION, a California Corporation,

Plaintiffs,

vs.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION, a Delaware Corporation, and AMERICAN AIR LINES, INC., a Delaware Corporation,

Defendants.

STIPULATION #7

It is hereby stipulated subject to proof of error that the appended are reproductions of printed publications and that said copies may be used in evidence with the same force and effect as an original, subject to any objection which may be made thereto as irrelevant or immaterial, when offered in evidence, viz:

"Exhibit 31" is a reproduction of page 140 No. 6 issued in the year 1937; "Exhibit 32" is a reproduction of page 419 No. 6 issued in the year 1937; "Exhibit 33" is a reproduction of page 609 No. 22 issued in the year 1937; "Exhibit 34" is a reproduction of page 421 No. 16 issued in the year 1938;

"Exhibit 35" is a reproduction of pages 144 and 145 No. 6 issued in the year 1939.

All are included in a periodical entitled "Flugsport" published and issued by "Flugsport" in Frankfurt, Germany, on said dates, respectively.

Exhibits 31a, 32a, 33a, 34a, 35a are translations of said articles respectively, subject to correction if any errors are found.

........................,
Attorneys for Plaintiffs.

/s/ ROBERT B. WATTS,
/s/ FRED GERLACH,
Attorneys for Defendants.

Exhibit 35a

Translation of page 144, No. 6—"Flugsport" (1939)

Performances and wing design of the DFS Reiher and DFS Weihe gliders were used in the construction of DFS Meise; for the root wing profile, Go 549 was thickened 16%; in the outboard wing Go 676 was used.

Illustration #1 shows that this profile is most suitable for the requirements of a compromise plane. The Ca region so important for this purpose is located between 0.6 and 1.4. Profile 549 is referred to twice in the series of experiments at Goettingen. As the coefficients disagree considerably, a third measurement has been undertaken by the DFS in a new larger tunnel at Goettingen—in illustration #1 marked III. A fourth comparative measurement should be based on the coefficients of profile Go 426.

This profile is identical with 549 with a slight change in thickness. For the purpose of comparison Profiles Go 532 and Go 535, well known in the construction of gliders as well as NACA 23012 are noted in the illustration; it was considered desirable to mathematically reduce the "Profile Resistance" of all these profiles to the thickness of Profile 549, Illustration I indicates that even if the most unfavorable units of measurement are used, the highest Ca is 0.6, Goettingen 549 is the best. Whether NACA 23012 is better for speed cannot be decided because of discrepancies in the measurements undertaken by DVL, compared to the measurements undertaken in the 7x10 tunnel and those in the American super-pressure tunnel. We shall have to wait for further measurements, possibly some taken in flight. On the other hand, it is a well-known fact, that Go 535 is the most favorable solution for slow flight.

In the outer panel of the wing from 0.6 of the semi-span Profile Go 676 has been used instead of 549; significant for Go 676 is the wide Ca range.

Admitted November 24, 1950.

DEFENDANTS' EXHIBIT BBB

District Court of the United States, Southern District of California, Central Division
Civil Action No. 10930-Y

MAURICE A. GARBELL, INC., a California Corporation, and GARBELL RESEARCH FOUNDATION, a California, Corporation,

Plaintiffs,

vs.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION, A DELAWARE CORPORATION, AND AMERICAN AIR LINES, INC., a Delaware Corporation,

Defendants.

STIPULATION #8

It is hereby stipulated subject to proof of error that the appended "Exhibit 36" is a reproduction of pages 355 to 356 Vol. XVIII fasc. 3 of a periodical entitled "L'Aerotecnica" issued and published by the Institute Poligrafico Dello Stato in Rome, Italy, during the year 1938, and that "Exhibit 36a" is a translation of said article (subject to correction if any error is contained therein) and that the said copy and translation may be used in evidence with the same force and effect as originals, subject to any objection which may be made thereto as irrelevant or immaterial, when offered in evidence.

LYON & LYON,
/s/ FREDERICK W. LYON,
Attorneys for Plaintiffs.

/s/ ROBERT B. WATTS,
/s/ FRED GERLACH,
Attorneys for Defendants.

Exhibit 36a

Translation L'Aerotecnica Vol. XDIII fasc. 3, 1938,
Pages 335 and 336
Figure 8

The "Asiago"
High Performance Glider (Italian)

Description:

Wing:

The "Asiago" has a high wing, of monospar construction, with one streamlined steel strut on each side. The wing spar is of the box type and made of laminated fir. The leading edge acts as a second spar to prevent wing torsion. Airfoils used are: G535 for the rectangular part of the wing, M6 for the tapering extremities. Transition from one airfoil to the other is linear.

The ailerons are rather big. The differential control has a ratio of 1:2.5. Ball bearings are used everywhere in the aileron controls. This makes for an extremely smooth lateral control of the airplane.

Almost all metallic parts are of national duraluminum.

To facilitate landings and flight in clouds, two slotted spoilers are mounted above the wings. With these open, rate of descent can be increased by more than 200 ft./min.

Fuselage:

The front part of the fuselage has a hexagonal section, rounded at the top, while the rear part is conical. The fuselage is of the hull type. The cockpit is very comfortable, having been designed for minimum pilot fatigue in flights of a long duration. The towing mechanism, which can be used for either winch launching or actual air towing, can be released through a small lever. The barograph is installed close to the pilot's head. The landing skid is robust and well suspended.

A tennis ball is used to absorb tail skid shocks.

The control stick is mounted on ball bearings.

Empennage:

Horizontal surfaces are cantilever. The stabilizer is attached to the fuselage by only four bolts. Controls are all inside the hull.

* * *

The "Asiago" has been built for maximum maneuverability, keeping in mind low cost and ease of construction. Imported materials represent a negligible portion of the total, as wide use has been made of fir, poplar, and dural, all available in Italy.

The "Asiago" has passed the tests of the "Acrobatic gliders" category.

Glider "Penguin G.P. 1"

The Penguin G.P. 1 is a glider of high efficiency built as a project of the Application Center of the Politechnic Institute, financed by the Institute. Vittorio Bonomi, well known glider pilot, and Angelo Ambrosini, Engineer, have collaborated in its construction.

General characteristics:

Wing Span	50 ft.
Length	21 ft. 4 in.

Wing Surface	164 sq. ft.
Aspect ratio	15
Weight Empty	375 lbs.
Useful Load	175 lbs.
Total Weight	550 lbs.
Wing Loading	3.1 lbs./sq. ft.
Strength Coefficient	9
Minimum Sinking Speed	136 ft./min.
Angle of Descent	1:25.3

Description:

Cantilever wing, with diehedral in the center section. This insures good stability and unobstructed visibility in all directions. Monospar wing—Airfoils G535 for the rectangular part of the wing, NACA 23012 for the tapered extremities. Transition between the two airfoils is linear. In the immediate vicinity of the fuselage, airfoil section G535 progressively becomes an NACA 0015. The transition is parabolic. The ailerons have a big surface, and there are two pairs, the outboard ailerons having a bigger displacement angle. This gives an excellent lateral control. Aileron control is through double differentials, ratio 1:2.5.

Admitted November 24, 1950.

DEFENDANTS' EXHIBIT CCC

District Court of the United States, Southern District of California, Central Division

Civil Action No. 10930-Y

MAURICE A. GARBELL, INC., a California Corporation, and GARBELL RESEARCH FOUNDATION, a California, Corporation,

Plaintiffs,

vs.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION, a Delaware Corporation, and AMERICAN AIR LINES, INC., a Delaware Corporation,

Defendants.

STIPULATION #9

It is hereby stipulated subject to proof of error that the appended are reproductions of printed publications and that said copies may be used in evidence with the same force and effect as originals, subject to any objection which may be made thereto as irrelevant or immaterial, when offered in evidence, viz;

"Exhibit 37" is a reproduction of page 116 of a printed text book entitled "Sailplanes" issued and published by Chapman Hall, Ltd., in London, England, during the year 1937; "Exhibit 38" is a reproduction of pages 80-81 from a printed text book entitled "Flight Without Power" issued and published by Pittman Publishing Corporation in New York, N. Y., during the year 1940; "Exhibit 39" is a reproduction of pages 128-129 from a printed text book entitled "First Flight Principles" issued and published by the American Technical Society in Chicago, Illinois, during the year 1941; "Exhibit 40" is a reproduction of page 69 from the printed text book entitled "Aircraft Design" Vol. 1, issued and published by Chapman and Hall in London, England, in the year 1938; "Exhibit 41" is a reproduction of pages 68, 69, 74, 75, 78, 79 and 92 of a publication of the "Flugtechnische Fachgruppe" issued and published by Technischen Hochchule of Aachen, Germany.

LYON & LYON,
/s/ FREDERICK W. LYON,
Attorneys for Plaintiffs.

/s/ ROBERT B. WATTS,
/s/ FRED GERLACH,
Attorneys for Defendants.

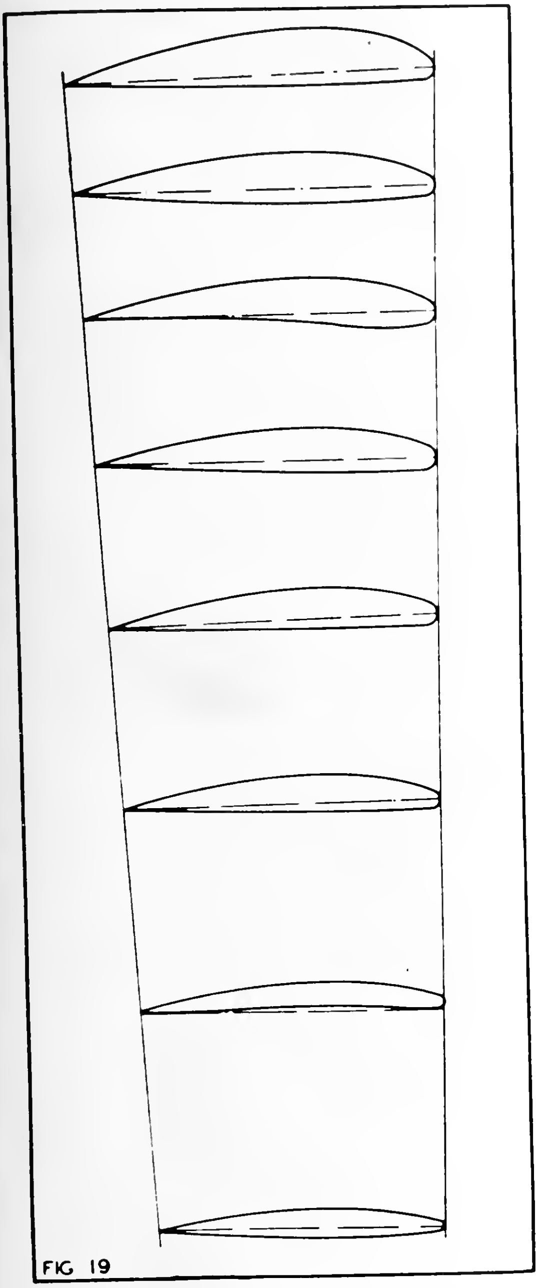


FIG 19

the L/D ratio at high angles of attack. This is in addition to its structural advantages which are great. Another may be mentioned. Satisfactory controlling effect may be obtained with smaller ailerons when the thicker sections are used in the wing.

Made possible by the development of the thick sections, the tapered wing has lately become very popular with designers. There are numerous reasons for this popularity. Some of these are structural and some are aerodynamic.

There are four ways in which the tapered wing may be constructed. The first and simplest of these is the tapered planform. Fig. 16. In such a wing, the greatest chord is the root chord. This lies nearest the fuselage. As the tip is approached, the chord decreases in length and with it all of the other dimensions of the section decrease in like ratio. If, for example, the chord at the tip is one-half of the chord at the root, the maximum upper ordinate at the tip is one-half of the maximum upper ordinate at the root and so on.

Fig. 17 shows another manner in which a tapered wing may be constructed. Here the planform is left rectangular and the thickness of the section is decreased as the tip is approached. The latter method is the same as multiplying all of the ordinates by some multiplier to obtain the ordinates for the section at each point. To show what is meant by this we choose another example. Suppose th[at] we have already decided the length [of] the chord. This will remain const[ant] throughout the span of the wing. At the tip we desire to have our wing only one-half as thick as it is at the root. We multiply all of the ordinates of the root section by .5 and the results will be the ordinates of the tip section. To obtain the ordinates of a section midway between the two, we multiply the ordinates of the root section by .75. There is thus established a relation between the location of a section and its ordinates. If we desire the ordinates of a section that lies midway between the

m: section already determined and the root, we seek a number which lies midway between .75 and 1. Obviously such a number is .875 and we obtain the ordinates of the desired section by multiplying the ordinates of the root section by .875. This will give the ordinates of the desired section. Similarly, to find the ordinates of a section that lies midway between the mid-section and the tip, we multiply the root ordinates by a number midway between .75 and .5 and this is, of course, .625. In this way we are able to construct a tapered wing having a variable section throughout its length. The section at any point, however, bears a simple relation to the section at the root.

There is a third way of tapering a wing that is a combination of t two mentioned. Fig. 18. Here the wing is tapered both in planform and in thickness. Certain advantages may be claimed for each of these types of tapered wings. A discussion of them does not properly come within the scope of this text. The individual prejudices of the designer are in many cases the determining factor in the selection of the type of taper used. Taper in thickness only is seldom used. Taper in planform is probably the most popular among designers.

There is another form of tapered wing in which the section at any point bears no simple relation to the root section. Fig. 19. According to this method, a section is selected which gives a satisfactory spar depth and satisfactory aerodynamic charac- istics for each point of the wing.

All of the advantages that are possessed by the thick wing sections are possessed by the tapered wing. In addition, there is a great decrease in the weight of the structure. This follows, because the bending moments and shearing stresses are greatest near the root of the wing, and the tapered construction allows the wing to be strongest at the points of greatest stress. It is an ideal construction for a monoplane because of the general cleanness of design that it permits. The entire wing structure is internal No external braces are required and the parasite drag is diminished by the amount of the drag of the eliminated external parts. Some disadvantage attends, however. The use of such a construction usually means an increase in structural weight. The addition of any weight to the structure diminishes by the same amount the useful load that the airplane will carry. This increase in weight, it must be remembered, applies as compared to airplanes that are constructed with wings of moderate thickness. As was mentioned earlier, the tapered wing, cantilever construction, allows a decrease in structural weight over that of the thick wing constant section construction. Hence, the thick sections are suitable only for root sections and are commonly so used.

Aspect Ratio

See Fig. 20. There is a dimension of the airfoil that is of considerable importance in performance. It is not a dimension of the section but is a dimension of the wing itself. We are interested in the effect that the shape of the wing has upon its characteristics. We are interested for the same reason that we were interested in the shape of the section. We desire to find a shape which will give us the maximum amount of lift with a minimum amount of drag.

come, to a large extent at least, by giving a twist to the wing, though theoretically the geometric twist necessary to produce an elliptical C_L distribution across the span is considerable, varying from about $-13°$ for a taper ratio of 2, to $-20°$ for a ratio of 5 (the twist should not be uniform, but should increase progressively towards the tip), and this in turn causes increased induced drag, the increase for a 20° twist being roughly 10 per cent. for a ratio of 2·5, and 20 per cent. when the taper ratio is 5. On account of this it is doubtful whether a twist greater than 3° or 4° should be used, and 2° might be regarded as a preferable limiting figure.

In practice a lesser amount of wash-out than the theoretical figures given above has been found necessary due to the presence of a fuselage, or engine nacelles, which have the effect of accelerating the advent of unstable, or stalled, air-flow over the inner part of a wing.

A better method of preventing tip-stalling, or one which may be profitably employed in conjunction with a small degree of twist, is to increase the camber from root to tip, or at least over the outer sections of the wing. Alternatively, the aerofoil section may be graded along the span so that the tip section has a greater angle of maximum lift, sufficient wash-out being employed to keep the angle of zero lift constant along the span. Increase of camber results in a greater angular range of lift,* i.e., from the no-lift angle to the angle for $C_{L\,max}$, the greater angle of the latter being made use of for delaying stalling towards the wing-tips. For taper ratios up to 4, a camber grading of from, say, 2 per cent. at the root to 5 or 6 per cent. at the tip is generally sufficient for satisfactory results.

Aerofoil sections with rearward position of maximum camber,† i.e., behind the one-third chord position, give better results than forward camber locations. Rearward shift of the point of maximum camber over the tip portion of a wing is likewise beneficial in this respect.

Another solution to the tip-stalling problem, and again one that may be used in conjunction with camber variation, is provided by suitable grading of the wing thickness over the outer portion of the span, but avoiding, if possible, the rather critical region of 12 per cent.‡

* See p. 58 (Chap. V).
† See pp. 59 and 60 (Chap. V).
‡ See p. 60 (Chap. V).

DEFENDANTS' EXHIBIT GGG

District Court of the United States, Southern District of California, Central Division

Civil Action No. 10930-Y

MAURICE A. GARBELL, INC., a California Corporation, and GARBELL RESEARCH FOUNDATION, a California, Corporation,

Plaintiffs,

vs.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION, a Delaware Corporation, and AMERICAN AIR LINES, INC., a Delaware Corporation,

Defendants.

STIPULATION #14

It is hereby stipulated subject to proof of error that the appended "Exhibit 128" is a reproduction of page 5 of the issue of February 5, 1938, of the printed publication "Le Vie Dell'Aria" containing an article entitled "Tre nuovi veleggiatori italiani" published and issued by Editorial Aeronautica in Milan, Italy, in the year 1938, and that "Exhibit 128a" is a translation of said article (subject to correction if any error is contained therein), and that said "Exhibit 128" may be used in evidence with the same force and effect as an original, subject to any objection which may be made thereto as

Defendants' Exhibit GGG—(Continued)

irrelevant or immaterial when offered in evidence, viz;

LYON & LYON,
/s/ FREDERICK W. LYON,
Attorneys for Plaintiffs.

/s/ FRED GERLACH,
/s/ ROBERT B. WATTS,
Attorneys for Defendants.

Exhibit 128a

Translation from Italian AJM:MS
Three New Italian Gliders

At the Arcore (Monza) Airport, where there took place the first flights of the one who is today the first aviator of Italy, there took place a few days ago the testing of three new gliders constructed during the last six months. The collaboration of the Center of Studies and Experiments for gliding of the Royal Polytechnical and of the GUF of Milan, on the one hand, and of the Aeronautica Lombarda, on the other hand, have resulted, with characteristic Fascistic rapidity, in a range of gliders which places Italy at the height of the most progressive countries, even with respect to gliding. There are not concerned planes constructed under license with foreign designs or copies like some planes which were made last year, but new models constructed on basis of the latest inventions and the latest Italian and foreign experience.

Each one of the three planes represents a stage in

Defendants' Exhibit GGG—(Continued)

the training of glider pilots of high class and in the sport development of future sport groups.

These are:

1. The "ASIAGO G.P. 2" designed by Garbell and Preti of the CVV, a glider for thermal soaring (C and D license).
2. The "ALCIONE B.S. 28" of Bonomi and Silva (Aeronautica Lombarda), an intermediate glider for high altitude gliding.
3. The "PINGUINO G.P. 1" of the CVV, a glider of the highest quality with which the college students of Milan will participate in the contests during the next season.

As we already announced last October in "Le Vie dell'Aria," the manufacturing program of the CVV is greatly assisted by the aeronautical fans Vittorio Bonomi and Eng. A. Ambrosini. The prototype of the CVV have been built by the shop-workers of the "Aeronautica Lombarda" and partly in the Cantu shop which up to the present time has supplied almost all the Italian elementary training gliders. At the same time the "Aeronautica Lombarda" has started the construction on a mass production basis of the models of the CVV, which has awakened general interest not only among the Italian glider pilots but also among the foreign pilots and organizations. Thus the collaboration between the CVV, a technical, scientific and sporting organization par excellence which must not and cannot attend to the mass production of its models, and the "Aeronautica Lombarda," a manufacturing plant of vast experi-

Defendants' Exhibit GGG—(Continued)

ence, was able to create in a short time these gliders for which the Italian pilots had been waiting.

The "Asiago G.P. 2" was born at the Aeronautical Exhibition of Milan. Thousands of visitors stopped in front of the stand of the RUNA and were present during the first stage of the assembling of this glider.

Here are its principal characteristic features:

Wing span 13.70 m.; length 6.50 m.; surface of the wings 12.70 m2; aspect ratio 14.8; weight without load 120 kg; useful load 90 kg; total weight 210 kg; wing loading 16.5 kg/m2 strength coefficient 9; minimum velocity of descent 0.80 m. per second; gliding angle 1:20.

The wing, of the mono-spar type with torsion-resisting leading edge, has a single profiled strut. For the purpose of good aero-dynamic efficiency and of low sucking speed of descent, there has been selected a comparatively large aspect ratio (14.8). The ailerons are very large (2.55 m2) and have a differential motion of a ratio of 1:2.5. On the upper side of the wing there is applied the well-known CVV flap which serves to increase, as may be desired by the pilot, the speed of descent of the apparatus, which is very necessary when landing outside of the aviation field and for flying into clouds. The CVV spoilers constitutes a simplified variant of the Jacobs spoiler (DFS).

The ample fuselage follows in general lines that of the "Anfibio Varese" of Rovesti-Mori. It is hexagonal (rounded) at the front part, and of rhombus

Defendants' Exhibit GGG—(Continued)

sections towards the tail. The pilot's seat is ample and commodious; it fits the shape of the body thus reducing to a maximum the fatigue of long flights. The cables of the pedal pass through the space between the double wall leaving the pilot's seat entirely free. The control stick is of duraluminum tubing so as not to affect the compass. All the controls move on ball bearings. Behind the head of the pilot between the fourth and fifth frames of the fuselage, there is a box for the recording barometer. Its cover serves at the same time as hand support. On the Asiago, the troublesome problem for the rest for the left hand has been solved. A simple but comfortable duraluminum rest finally assures the pilot the desired rest for his left hand. Near this rest are located the levers for the operation of the flaps and for the releases. The two releases—the open one for winch launching and the closed one for the air drag—are simultaneously opened with a single handle.

The horizontal empennage is of the cantilever type and is attached to the fuselage by means of three bolts, in addition to the inside control bolt. The rudder is low and of modern lines.

The greatest attention has been given to obtaining ease of operation in the controls, in connection with which up to the present time, many gliders used to leave a great deal to be desired. As is universally known, the sensitivity of the elevator of a glider is equal to, if not superior than, than that of a motor airplane; the ailerons are already more inert, but worst of all is the rudder which generally

Defendants' Exhibit GGG—(Continued)

has very little effect. In the Asiago, the ailerons are very efficient, this being due to selected profiles (G 535 and NACA M6) and also due to the aerodynamic wing warp. The rudder on the other hand has been placed behind the elevator in order to increase the arm and therefore the momentum. The apparatus responds very well to the controls. Someone who perhaps exaggerates states that "it is just like a CR." This arrangement has the advantage of also avoiding interference between the horizontal and vertical empennages during spinning, as was discovered a few months ago by the Zurich scientist Haller.

The landing members are the following: a standard front skid and a small tail skid made resilient by means of a tennis ball.

In the construction of the "Asiago," considerable use was made of material produced in Italy (fir, poplar, duraluminum).

The "Aeronautica Lombarda" is now manufacturing the "Asiago" on a large production basis—which plane, due to its simplicity, can be sold at comparatively low price—which, in addition to the surprising flight qualities which are superior to all the Italian and foreign planes manufactured up to the present time, will greatly favor its diffusion.

The "Alcione B.S. 28" of Engineer Camilla Silva is endeavoring to meet the need felt by the schools for high altitude gliding, which desired a comparatively economical plane which still had flying qualities like those of the large gliders in order to im-

Defendants' Exhibit GGG—(Continued)

prove the training of pilots who have already completed their training in the gliding school.

Here are the technical specifications of this glider: wing span 14.50 meters; length 6.55 meters; area of the wings 14 square meters; aspect ratio 15; weight without load 160 kgs; useful load 85 kgs; total weight 245 kgs; wing loading 17.5 kg/m2; strength coefficient 9; minimum velocity of descent 0.75 m/sec.; gliding angle 1:22.

The "Alcione" is provided with a middle wing, full cantilever, straight, and of a fully tapered plan.

The profiles used are G449, G693, NACA 23012, NACA 0012.

The entire trailing edge is occupied by movable surfaces. The inside third forms the camber flaps controlled by a lever located on the left side of the pilot. The other 2/3, the "ailerons," are divided into halves and are controlled with double differential. In addition to the differential motion between the right-hand aileron and the left-hand aileron, the outer aileron has a greater amplitude than the inside one and this motion approximates the warp of the wings of birds, thus improving the transverse maneuverability.

On the upper side of the wing, there is located the CVV flap.

The fuselage is of hexagonal section with rounded upper part. The tail surfaces correspond to those of the Asiago.

In addition to the main skid, there is a small cen-

Defendants' Exhibit GGG—(Continued)

tral wheel which facilitates the landing and the take-off.

* * *

Finally, the "Pinguino G.P. 1" has all the characteristic features of a large glider: middle M wing, rounded fuselage, very accurate connections. Its construction was made possible by the generosity of the Royal Polytechnical and of the well-known glider pioneer, Vittorio Bonnomi. Here are its characteristic features:

Wing span 15.30 meters; length 6.50 meters; wing surface 15.20 square meters; aspect ratio 15; weight without load 170 kgs.; useful load 80 kgs. total weight 250 kgs.; wing loading 15.2 kg/m2; strength coefficient 9; minimum descending velocity 0.69 m/sec.; gliding angle 1:25.3.

The wing is a full cantilever and has a dihedral angle of 6° at the central part. The profiles used are the G 535 and the NACA 23012, with aerodynamic warping of about 3°. The wing is of the single spar type. The ailerons are very large and have a strongly differential control. Also here, the CVV flaps are not missing.

The fuselage is of ovoid section. Special care was given to the connection between the wing and the fuselage.

* * *

The excellent flying qualities of these three new gliders have been shown by tests carried out on January 29th and 30th last, at the Arcore Airdrome by the Engineer, Colonel Nannini and by the In-

Defendants' Exhibit GGG—(Continued)

structor Aldo Tavazza, who, with the gliders, carried out some stunts, after having been released from the tugging plane at a height of 1,000 meters. There were present at the tests of the new gliders: Prof. Cassinis, President of the E. De Amicis Study Center; Engineer Silva of the Aeronautica Lombarda for gliding; Instructor Plinio Rovesti of Varese, and Engineer Bracale of the Aeronautic Registry.

The flight tests have fully confirmed the maneuverability and stability of the new gliders and will soon be followed by soaring tests. It must be noted that in the afternoon of the 30th, the Pilot Venturini effected, with the "Asiago G.P. 2," a series of stunts which were perfectly successful in view of the trim compensation of the glider. The pilot, who is a holder of a "C" flying license and of a first-grade airplane license, had not up to that time done any stunt-flying.

MAURIZIO GARBELL.

Caption Under Illustration:

Top: The "Alcione B.S. 28" taking off under the pull of the winch. There can be noticed the low camber flaps.

Bottom: The "Pinguino G.P." in full flight.

Admitted November 24, 1950.

DEFENDANTS' EXHIBIT HHH

District Court of the United States, Southern District of California, Central Division

Civil Action No. 10930-Y

MAURICE A. GARBELL, INC., a California Corporation, and GARBELL RESEARCH FOUNDATION, a California, Corporation,

Plaintiffs,

vs.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION, a Delaware Corporation, and AMERICAN AIR LINES, INC., a Delaware Corporation,

Defendants.

STIPULATION #15

It is hereby stipulated subject to proof of error that the appended "Exhibit 129" is a reproduction of pages 58 and 59 of Issue No. 3, February 2, 1938, of the printed publication "Flugsport" containing an article entitled "Leistungssegler 'Pinguino G.P. 1' " published and issued by Flugsport in Frankfurt, Germany, in 1938, and that "Exhibit 129a" is a translation of said article (subject to correction if any error is contained therein), and that said "Exhibit 129" may be used in evidence with the same force and effect as an original, subject to any objection which may be made thereto as irrelevant or immaterial when offered in evidence, viz;

LYON & LYON,
/s/ FREDERICK W. LYON,
Attorneys for Plaintiffs.

/s/ FRED GERLACH,
/s/ ROBERT B. WATTS,
Attorneys for Defendants.

Exhibit 129a

Translation from "Flugsport" Feb. 2, 1938
No. 3, p. 58-59

High-Performance Glider "Pinguino G.P. 1"

The glider was built in the second half of the year 1937 by students of the Milan Technical College, with financial aid from the College, from the noted advocate for gliding flight in Italy: Vittorio Bonomi, and from the aircraft industrialist Angelo Ambrosini. The design for the machine came from Garbell and Preti, of the CVV (Centro Studi ed Esperienze per il Volo a Vela).

The "Pinguino" is constructed as a mid-wing cantilever with a gull wing. Single-spar construction; with plywood nose. Profile to the bend, Gottingen 535; from here outward it merges linearly into the NACA 23012 section. At the transition from the wing to the fuselage the G 535 wing section runs into the NACA 0015 fuselage profile. Big ailerons; double differential control, with an angular deflection ratio of 1:25. The outer halves of the divided ailerons are more deflected, whereby a considerable improvement of their action is obtained. A further advantage consists in that, during bending of the wing, no binding of the ailerons occurs. (An old and typical example of this arrangement is the Russian long-distance craft "Ant. 25," whose ailerons are subdivided into four single flaps). For the purpose of increasing the rate of descent at will, two CVV spoiler flaps are installed on the suction side, which, in the manner of the braker flaps developed by DFS (see "Flugsport" of 1937, page 350), may be deflected forward on a circular arc, and thereby leave a gap open between the lower edge of the flap and the suction side of the wing.

Fuselage of oval cross section, coming to an edge underneath. Comfortable pilot's seat. Instruments fastened to the fuselage itself, not to the cowling, in order that the cowling may not become too heavy, and possibly hinder rapid emergence when there is danger. The cowling is held in place by a DFS speed catch. The release lever simultaneously operates the open winch hook and DFS tow coupling, which are prescribed in Italy.

Wing span, 15.3 m; length, 6.5 m; area 15.2 m2; aspect ratio, 1.15; empty weight, 170 kg; load, 80 kg; flying weight, 250 kg; wing loading, 15.2 kg/m2; breaking load factor in case A, 9; minimum rate of descent, 0.69 m/sec.; maximum drag/lift ratio 1:25.3.

Translated by W. G. Weekley.

Admitted November 24, 1950.

DEFENDANTS' EXHIBIT III

District Court of the United States, Southern District of California, Central Division

Civil Action No. 10930-Y

MAURICE A. GARBELL, INC., a California Corporation, and GARBELL RESEARCH FOUNDATION, a California, Corporation,

Plaintiffs,

vs.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION, a Delaware Corporation, and AMERICAN AIR LINES, INC., a Delaware Corporation,

Defendants.

STIPULATION #16

It is hereby stipulated subject to proof of error that the appended "Exhibit 130" is a reproduction of pages 538 and 539 of Issue No. 20 of September 29, 1937, of the printed publication "Flugsport" published and issued by "Flugsport" in Frankfurt, Germany, in the year 1937, and that "Exhibit 130a" is a translation of a part of said article (subject to correction if any error is contained therein), and that said "Exhibit 130" may be used in evidence with the same force and effect as an original, subject to any objection which may be made thereto as irrelevant or immaterial when offered in evidence, viz;

LYON & LYON,
/s/ FREDERICK W. LYON,
Attorneys for Plaintiffs.

/s/ FRED GERLACH,
/s/ ROBERT B. WATTS,
Attorneys for Defendants.

Exhibit 130a

Translation from "Flugsport" No. 20
Sept. 29, 1937, p. 538-9

The "Centro Studi ed Esperience per il Volo a Vela" (Testing Station for Gliding Flight) of the Milan Royal Technical College exhibits the construction of the training glider "Asiago G.P. 2" on the stand of the National Royal Aeronautical Club.

The machine is the result of experience in the Asiago Glider School, and is intended for training in thermal current and cloud flying. The designers Garbell and Preti also made use of their experience with the "Grunau-Baby" of the Polish Komar, and with the "H 17." As respects cloud flight, the machine has a load factor of 9, and air brakes on the upper side of the wing.

Profiles G 535 and M 6, with gradual transition. Most of the covering is pure Italian dural. In addition, all weak points that may be stressed use popular plywood instead of northern birch plywood. The fuselage is hexagonal in front, with a rounded cowling, and merges behind into a rectangular section.

Wing span, 13.7 m; length, 6.5 m; area, 12.7 m2; aspect ratio, 1:14.8; empty weight, 120 kg; flying weight, 210 kg; wing loading 16.5 kg/m2; drag/lift ratio, 1:20; rate of descent, 80 cm/sec.

The "Pinguino G.P. 1" machine built last summer by the same designers, and which belongs to the Sperber class, could not be exhibited for lack of space. This machine was built merely for study, and will therefore not go into production. Directly after the exhibition, this machine, which has an interesting choice of profiles (NACA 0015, G 535, NACA 23012), will be subjected to thorough tests at the Sezze-Littoria (Agro Pontino) fields.

In our next number, we shall report in detail about the Aeronautica Lombardia, the successor to Aeronautica Bonomi, company's mid-wing "Alcione B.S. 28" designed by Silva.

Translated by W. G. Weekley.

Admitted November 24, 1950.

DEFENDANTS' EXHIBIT JJJ

District Court of the United States, Southern District of California, Central Division

Civil Action No. 10930-Y

MAURICE A. GARBELL, INC., a California Corporation, and GARBELL RESEARCH FOUNDATION, a California, Corporation,

Plaintiffs,

vs.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION, a Delaware Corporation, and AMERICAN AIR LINES, INC., a Delaware Corporation,

Defendants.

STIPULATION #17

It is hereby stipulated subject to proof of error that the appended "Exhibit 131" is a reproduction of page 5 of the Issue of October 16, 1937, of the printed publication "Le Vie Dell'Aria" published and issued by Editorial Aeronautica in Milan, Italy, and that "Exhibit 131a" is a translation of the article "Il Volo A Vela" (subject to correction if any error is contained therein), and that said "Exhibit 131" may be used in evidence with same force and effect as an original, subject to any objection which may be made thereto as irrelevant or immaterial when offered in evidence, viz;

LYON & LYON,
/s/ FREDERICK W. LYON,
Attorneys for Plaintiffs.

Defendants' Exhibit JJJ—(Continued)

/s/ FRED GERLACH,

/s/ ROBERT B. WATTS,

Attorneys for Defendants.

Exhibit 131a

Translation from Italian WB:FG

At the Milan Salon

Gliding

While at the first Aeronautical Salon of Milan gliding occupied a very modest position, at the recent Salon it has assumed the position and importance due it.

In accordance with the very great value attributed to gliding by the Germans, the German representative made a large contribution to this exhibition. The Minister of Aeronautics of Berlin presented at different stands, the technical and sporting results of his organizations. The N.S.F.K. (National Socialist Flying Corps) showed by various graphs, models, etc., their work in the field of gliding and aeronautical craftsmanship: about 20 regular gliding schools, about 200 gliding groups and a large number of schools in aeronautical construction are preparing future pilots and skilled workers for the Air Force and civilian aviation.

Among the gliding schools, there are some which give excellent instruction in blind flying, instrument navigation and aerobatics. The records attained by German gliding are: 41 hours flight, 4650 meters altitude and 504 kilometers distance. These figures confirm, even numerically, the great stage of development obtained by this branch of aeronautics.

Defendants' Exhibit JJJ—(Continued)

The DUL Institute for aeronautical research is exhibiting a vertical wind tunnel for spinning tests constructed by the DFS (German Glider Research Institute).

The DFS is exhibiting, in its own stands, extremely interesting scientific material. In addition to the diagrams and photographs on weather study, which have been of such vital importance in the history of gliding, the DFS is exhibiting its own two aerodynamic smoke tunnels. By introducing into the current of air, thin smoke filaments, an attempt is made to study the very minute aerodynamic problems which ordinary aerodynamics cannot solve. In particular, the action due to the moving parts of the wing, such as flaps, flip increasors, air brakes, etc., can be evaluated with clearer precision. The DFS has developed, under the supervision of Alessandro Lippisch, two types of smoke tunnels, one economical, low priced type for elementary demonstration purposes for glider schools and a more involved type for scientific investigation. Particularly the second type has found great appreciation on the part of the representatives of the leading Italian scientific institutions.

The same Institute is also exhibiting a series of models of the main planes created by Engineers Lippisch and Jacobs.

The German glider industry has sent two of its best representatives, Hirth and Schweper. Hirth has brought to the exhibit his new two-seater Minimoa 2, a real masterpiece of precision and design. The Minimoa Goppingen 3, the Goppingen 4, a two-

Defendants' Exhibit JJJ—(Continued)

seater with the seats arranged alongside each other for instruction purposes, and finally the Wolf, one of which is being tested at present at the glider school established at Sezze by the R.U.N.A., are also extremely interesting.

Schweyer however is exhibiting one of its most characteristic constructions, the "Habicht" plane for aerobatics, designed by Jacobs of the DFS. Of a structure similar to the Rhonsperber, which plane is designed for a speed officially measured at 400 kilometers per hour; however, piloted by the "Commander of Glider Pilots," Hanna Reitsch, it has already repeatedly obtained speeds of more than 450 km. per hour. Its amazing ease of handling makes it possible to effect practically any stunt maneuver. The Italian Olympic squadron had an opportunity to see, in Berlin, the stunts of Hannah, among them front loops with two barrels while ascending, etc. The glider which was finally acquired also by the French champion, Marcel Thoret, is of beautiful mechanical and structural design. Also the flap mechanism, the ailerons, rudders and elevators are of a perfection rarely found in aeronautical construction, but rather found in optical and electrical apparatus.

The Habicht, together with the famous "slow" airplane Storch, had already the very high honor of being thoroughly inspected by the Duce during his stay in Germany at the Rechlin Camp.

Schweyer also exhibits the usual models of planes constructed by it, such as the two-seater Kranich,

Defendants' Exhibit JJJ—(Continued)

the Rhonsperber and a few other types of lesser importance.

To this group of German exhibitors, rich in more than 15 years of experience, there is added a small but courageous Italian representation.

The Aeronautica Lombarda (formerly the Aeronautica Vittorio Bonomi) presents the BS 28 designed by Engineer Silva. There is concerned a glider with middle wing of 14.50 meters wing spar and an aspect ratio of 15 meters. Through a special selection of the profiles (G. 449, G. 693, NACA 23012 and NACA 0012) there has been obtained a very fine wing and at the same time a wing sufficiently rigid and light in weight. Along the trailing edges of the wing, camber flaps and the four ailerons follow each other. The ailerons are actuated by means of a special differential control so as to assure a greater amplitude of motion the outer ailerons than of the inner ailerons, thus obtaining a greater ease of handling. Also, the differential ratio between the two pairs of ailerons is rather high and reaches a value of 1:2.5. The plane is provided with flaps. The cabin is designed with special regard to visibility in all directions. As landing members there have been installed a skid and a single wheel undercarriage. The cowling covers not only the cockpit but also the junction of the wings. Upon removing same, everything is uncovered, which greatly facilitates the assembling of the plane. The wings are connected with each other by means of connections of duraluminum, while the fuselage is connected by means of only four bolts inasmuch

Defendants' Exhibit JJJ—(Continued)

as it does not have to support any bending force. In general, practically all the metal parts are of duraluminum. In all the points where the stresses are less, use is made of poplar plywood instead of birch plywood, and poplar and fir are used instead of spruce.

The Center for Gliding Studies and Experiments of the Royal Polytechnical is exhibiting at the stand kindly placed at its disposal by the R.U.N.A., a model shop, which, constructed during the period of the exhibition, the model of the "Asiago" glider the design for which had been made by the Milanese students, Preti and Garbell.

This Center organized in 1934 by the late Liberato De Amici was able, with the assistance of the Royal Polytechnical, to gradually develop during the last two years. The interest taken in same by Knight Commander Bonomi and by Engineer Amorosini and also the assistance of the aeronautical authorities have made it possible to continuously increase the work. Thus the meteorological section was able to organize, in agreement with the Ministry of Aeronautics, a weather study department, and lately also the model section has started a promising activity. Already this summer there has been built the "Pinguino G.P. 1" which is now being perfected. There has now been created the G.P. 2 "Asiago" which was actually constructed at the exhibition. There is concerned a training plane for students who wish to qualify for a C and D license for thermal gliding and gliding in the clouds. The entire project was carried out with the intention

Defendants' Exhibit JJJ—(Continued)

of reducing as much as possible the moments of inertia around the vertical and longitudinal axes in order to obtain a maneuverability of the aileron and foot (base?) harmonized with the always somewhat excessive maneuverability of the elevator. The experience obtained from the Asiago glider contest has brought about the development of the CVV (Gliding Center) flap which is not to be used for landing, which is already very easy with a glider of the Asiago kind, but is to permit students to avoid entering the clouds. While today a student who is drawn in by a thick cloud tries vainly to avoid zooming up into the cloud, the CVV flap has for its purpose increasing, without any strain on the plain, the speed of descent by at least 1 meter per second. Thus there is obtained a further safety device which will be appreciated both by the students and their instructors.

What we have already stated in connection with the BS 28 also applies to the use of independent equipment. The Asiago plane will be reproduced by the Aeronautica Lombarda, which has taken a special interest in the development of the model and of the future mass production of same. The plane, as also the BS 28 and the Pinguino, will pass the official gliding tests at the experimental field of Agro Pontino.

The Italian gliding exhibition finally includes a beautiful collection of the GUF of Rome which shows some fine projects of gliders and sailplanes in addition to an engine plane of Fidia Piatelli.

Gliding is obtaining the position it deserves also

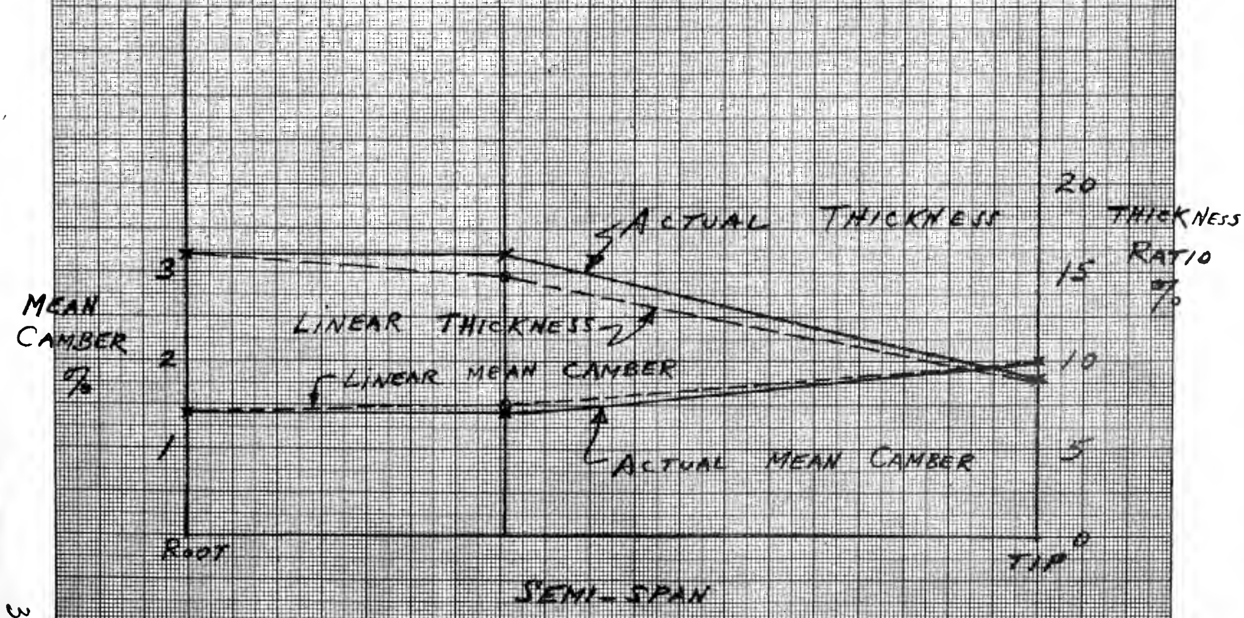
Defendants' Exhibit JJJ—(Continued)

among us, as was the case in other countries. As His Excellency Valle stated at the Asiago contest, we have made great progress also in gliding in achieving the standing to which Italian aviation has arisen.

MAURIZIO GARBELL.

Admitted November 24, 1950.

Case No. 10930-Y
CARSELL vs CONSOLIDATED
DFT EXHIBIT LLL
Date ______ No ______ IDENTIFICATION
Date NOV 24 1950 No LLL IN EVIDENCE
Clerk, U. S. District Court, Sou. Dist. of Calif.
John A. Childress Deputy Clerk



345

[illegible]

ENGINEERING REPORT

CURTISS-WRIGHT CORPORATION

SAINT LOUIS AIRPLANE DIVISION

ROBERTSON, MISSOURI

TITLE: AERODYNAMICS REPORT

AUTHOR: W.A.Sangster & W.T.Butterworth

Two-Place Basic Combat, Model 23
Engine - P & W Wasp, S3H1
Circular Proposal No. 39-100
U.S.Army Type Specification No. C-901

Approval [illegible] Ch.Engr.

DATE: February 20, 1939 APPROVALS: [illegible] Proj.Engr

REVISIONS

CHANGE LETTER	DATE	DESCRIPTION	AUTHOR	APPROVAL	NO. OF PAGES 109

DEFENDANTS'

EXHIBIT 136

Physical Characteristics

a. Design gross weight 4600 lbs. (Ref. 5)

b. Fuel: Normal 104 gals. 624 lbs. (Ref. 5)
Maximum 140 gals. 840 lbs. (Ref. 5)

c. Wing Dimensions
I - Airfoil section designation:
Root: CW-23 (at centerline)
Wing Splice: NACA 2314 (55.6 ins. outboard)
Tip: CW-19 (15 ins. from tip)

II Total supporting surface area: 174.3 sq.ft.

III Incidence: + 1° (Chord plane relative to Thrust Line)

IV Dihedral: 5.50° on Chord plane.

V Sweepback: Trailing edge normal to centerline of Airplane

d. Horizontal Tail Surfaces:

I	Total Area	25.56 sq.ft.
	(Including Blanketed Area	28.14 sq.ft.
II	Span	11.0 ft.
III	Maximum Chord	3.76 ft.
IV	Distance from Normal c.g. to 1/3 Maximum Chord point	184.65 in.
V	Stabilizer Area	16.98 sq.ft.
	Normal position relative to Thrust line 30°	
VI	Elevator Area (including tab)	8.58 sq.ft.
	2 Trim tabs, area each	.35 sq.ft.

e. Flaps

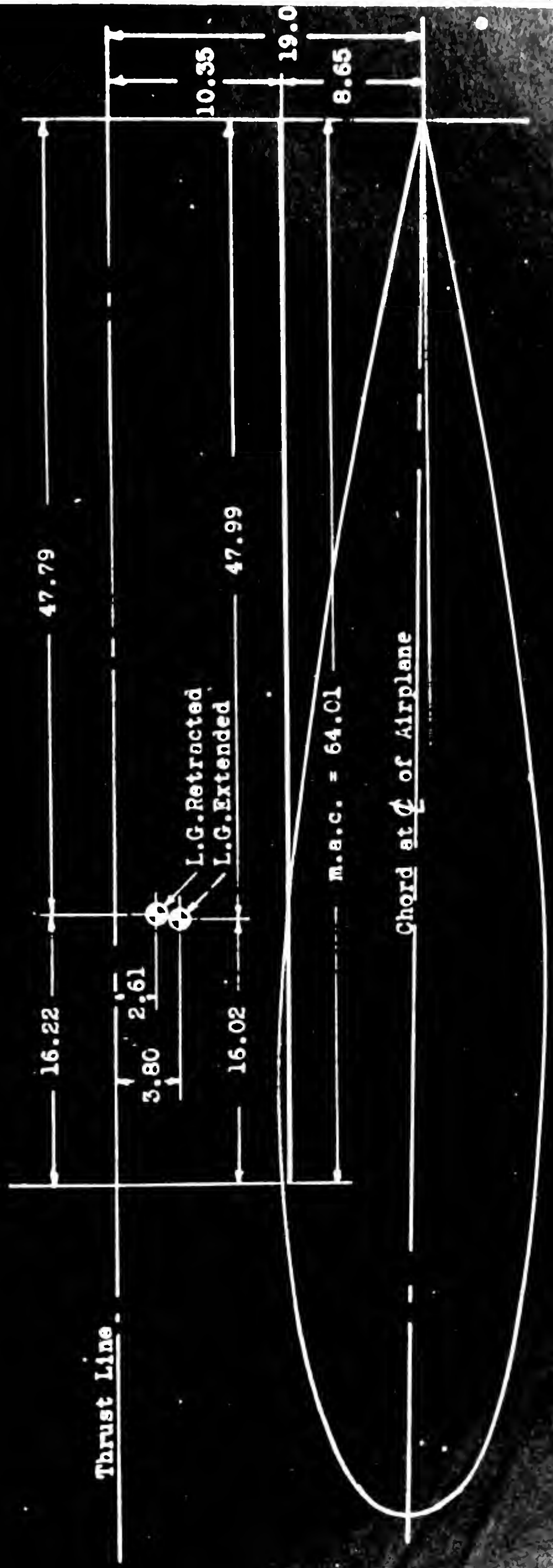
I The wing is equipped with split flaps over the central portion of the span. The flap extends over the rear 15% of the wing chord and is hinged along its forward edge.

II The dimensions and location of the flap are shown on the wing drawing, page 19 fig. 3.

Ratio (flap chord)/(wing chord) = .15
Ratio (flap span)/(wing span) = [illegible]
Total flap area = [illegible]

Fig. 4

AIRPLANE BALANCE



b_1 = span of center panel = 55.6"
b_2 = span of outer panel = 149.4"
A_{tot} = Total wing area = 174.3 sq.ft.

The mean slope of the lift curve, moment coefficient about the aerodynamic center, and location of the aerodynamic center are determined in the following calculations. Tables 3 and 4 present the average values of drag coefficient and angle of attack for the complete range of lift coefficients.

Aerodynamic Section Properties (Ref. Fig. 6)

Item	C.W.-23	NACA 2314	C-W-19
m = Slope of lift curve for aspect ratio 7.03	.0801	.0805	.0856
C_m = Moment Coefficient about aerodynamic center	-.002	-.035	-.055
a.c. = Aerodynamic center in fractions of chord	.232	.245	.236

$$m_{av} = \frac{[(.0801)(84.00) + (.0805)(71.45)]\,55.6 + [(.0805)(71.45) + (.0856)(38.76)] \times 149.4}{144(174.3)}$$

$$m_{av} = \underline{.0816/\text{deg.}}$$

$$C_{m_{av}} = -\frac{[(.002)(84.00)+(.035)(71.45)]\,55.6 + [(.035)(71.45)+(.055)(38.76)]\,149.4}{(144)(174.3)}$$

$$C_{m_{av}} = \underline{-.0335}$$

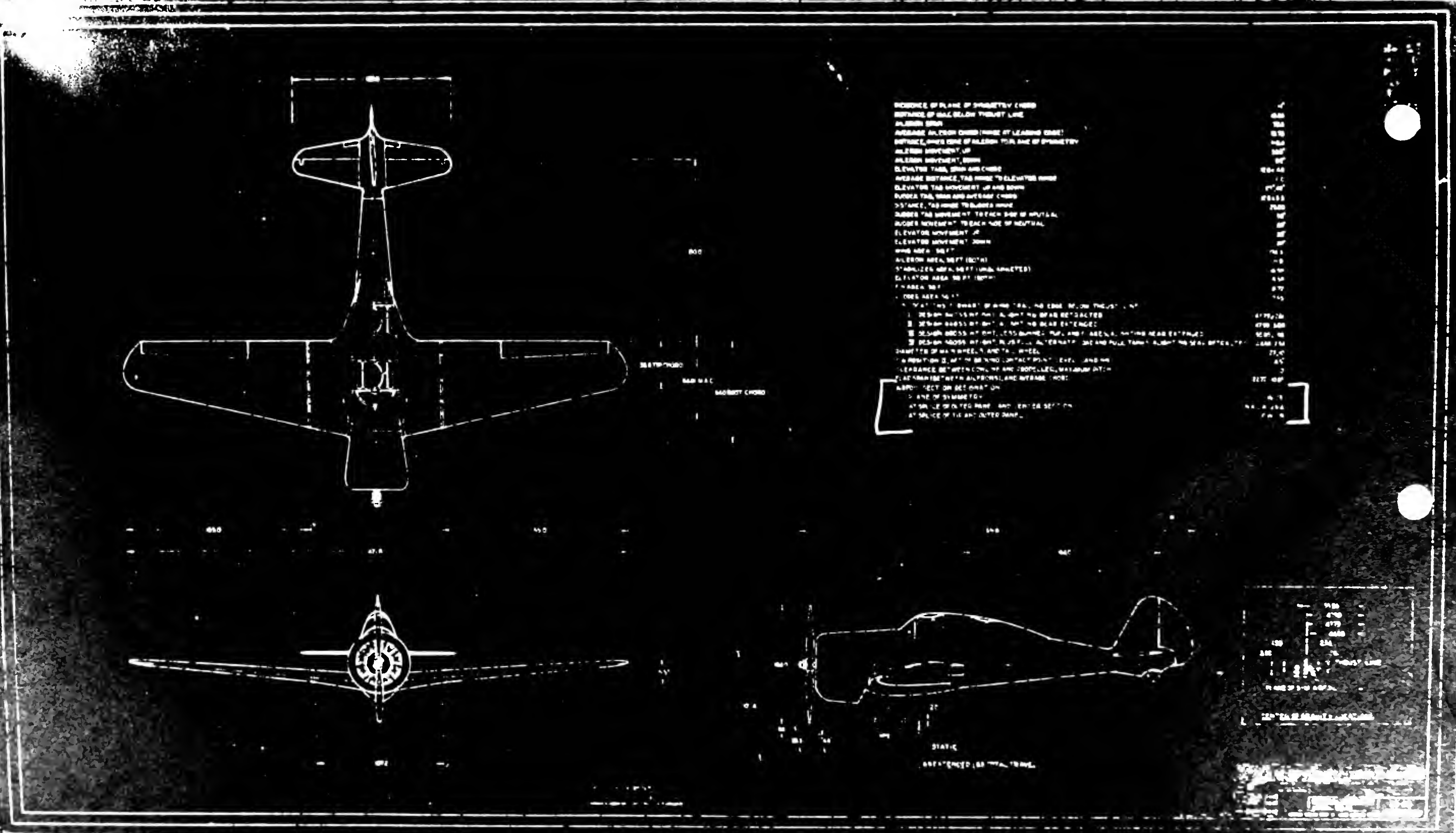
$$(a.c.)_{av} = \frac{[(.232)(84.00)+(.245)(71.45)]\,55.6 + [(.245)(71.45)+(.236)(38.76)]\,149.4}{(144)(174.3)}$$

$$(a.c.)_{av} = \underline{.241}$$

The wing normal and chord force coefficients for the complete range of lift coefficients of the mean wing alone are computed in Table 5. The accompanying pitching moment, tail, and airplane force coefficients are computed in succeeding Tables for various specific loading and flight conditions of the airplane.

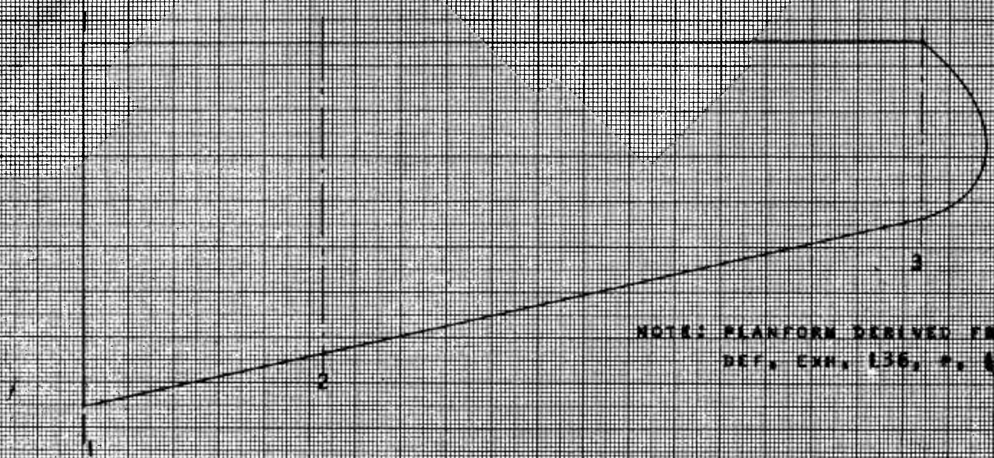
The airfoil section at the centerline (rib 1) is a symmetrical section, generated in the following manner: The upper contour of an N.A.C.A. 2315 airfoil section was reflected about the geometric chord line to form a symmetrical airfoil of 19% thickness lying in a plane normal to the plane of the chords. This resulting contour is designated as the CW-23 airfoil section, and to obtain the aerodynamic characteristics of that section, it was assumed to be equivalent to an N.A.C.A. 0019 airfoil section. The contours of the CW-23 airfoil section at rib 1, and of the N.A.C.A. 2314 airfoil section at rib 4 are obtainable from published N.A.C.A. airfoil data. The contour and table of ordinates of the CW-19 airfoil section is given on page 28a. All sections are taken in a plane normal to the plane of the chords.


A
B
C
55.6
13.05
8.84
214.2
3


STATIC

Case No. 10430-Y
Marbell Inc vs Consol Vultee et al
Dfts EXHIBIT NNN
Date NOV 24 1950 No. NNN IDENTIFICATION
Date NOV 24 1950 No. NNN IN EVIDENCE
Clerk, U. S. District Court, Sou. Dist. of Calif.
By [signature] Deputy Clerk

CURTISS-WRIGHT MODELS 21B AND 23
(REFER: DEF. EXH. 136)

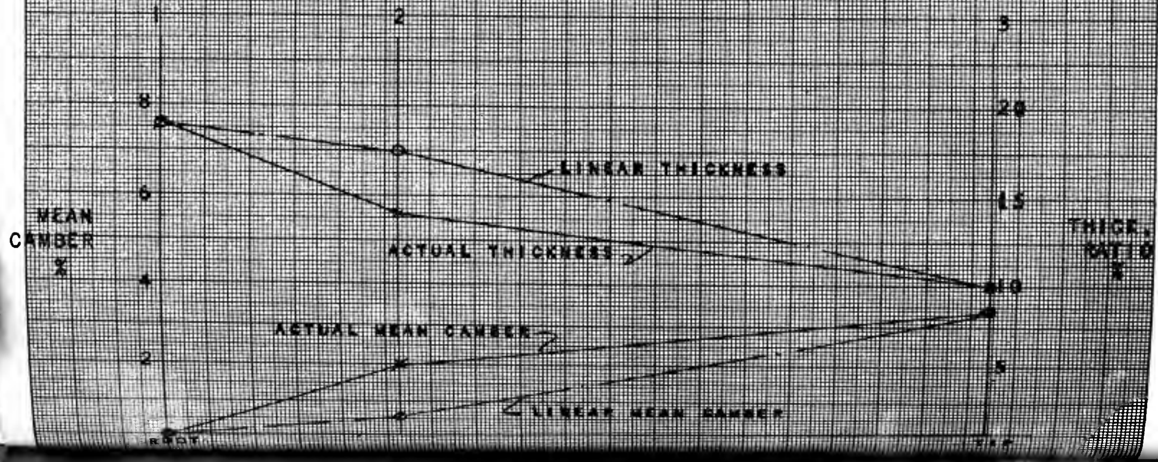


NOTE: PLANFORM DERIVED FROM DEF. EXH. 136, P. 69

SECTION	1	2	3
PROFILE	CW 23	NACA 2314	CW 19
LOCATION % OF SEMI-SPAN	0	26.8	92.9
MEAN CAMBER % OF CHORD	0	2.0	3.4
THICKNESS RATIO % OF CHORD	19.0	14.0	10.0

LINEAR DISTRIBUTION

MEAN CAMBER % OF CHORD	0	.56	3.4
THICKNESS RATIO % OF CHORD	19.0	17.5	10.0



DEFENDANTS' EXHIBIT OOO

Report No. 19-C4

Curtiss-Wright Airplane Co.

Robertson, Mo.

Engineering Department

Curtiss-Wright Sparrow, Model 19L
(2PCLM)

1 Lambert R-266, 90 H. P. Engine

Modification of Wing
Structural Considerations

Submitted By
/s/ LLOYD F. ENGELHARDT

Section....Structures.

No. of Pages: 11. Date: Oct. 3, 1935.

Revisions ....

Pages...........
Date Affected Remarks

....

....

Modification of Wing

Introduction:

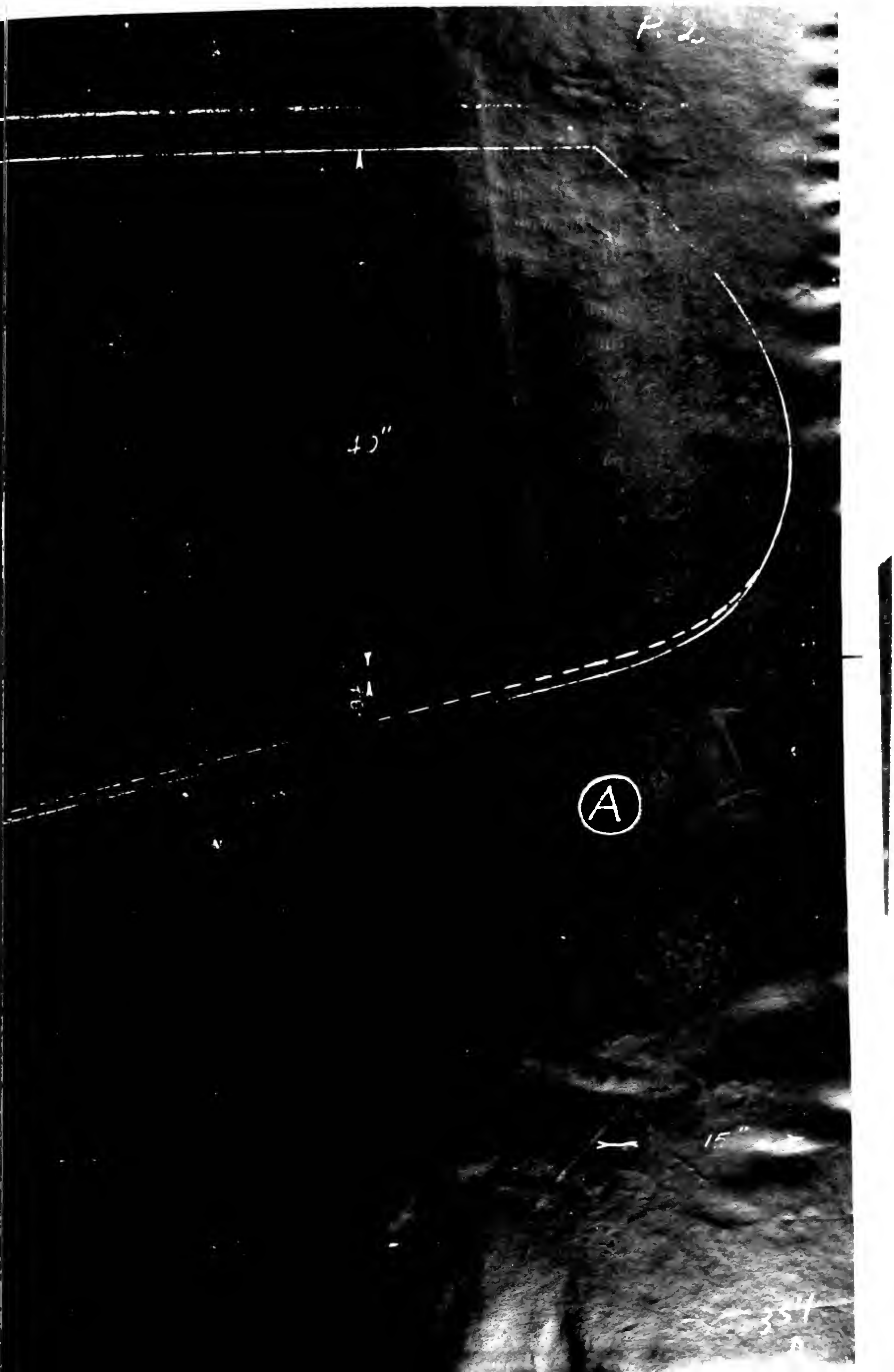
In order to meet the especially rigid requirements of a particular customer relative to stalling and spinning it was found necessary to modify the contour of the airfoil section in the outer portion

Defendants' Exhibit OOO—(Continued)

of the wing and at the same time to give the wing tip a certain amount of "wash-out."

To accomplish these changes an outer shell is added which extends below and slightly forward of the original lines. The plan form on Page 2 shows the change in area. The airfoil section on Page 3 shows the modification of the airfoil at the station 15 inches inboard of the tip (designated as A on page 2). On page 4 is a foreshortened plot of the mean camber lines of the modified airfoil and several of the N.A.C.A. series.

Drawing No. 19-03-13 gives all details of construction of the outer shell addition.


A
15"

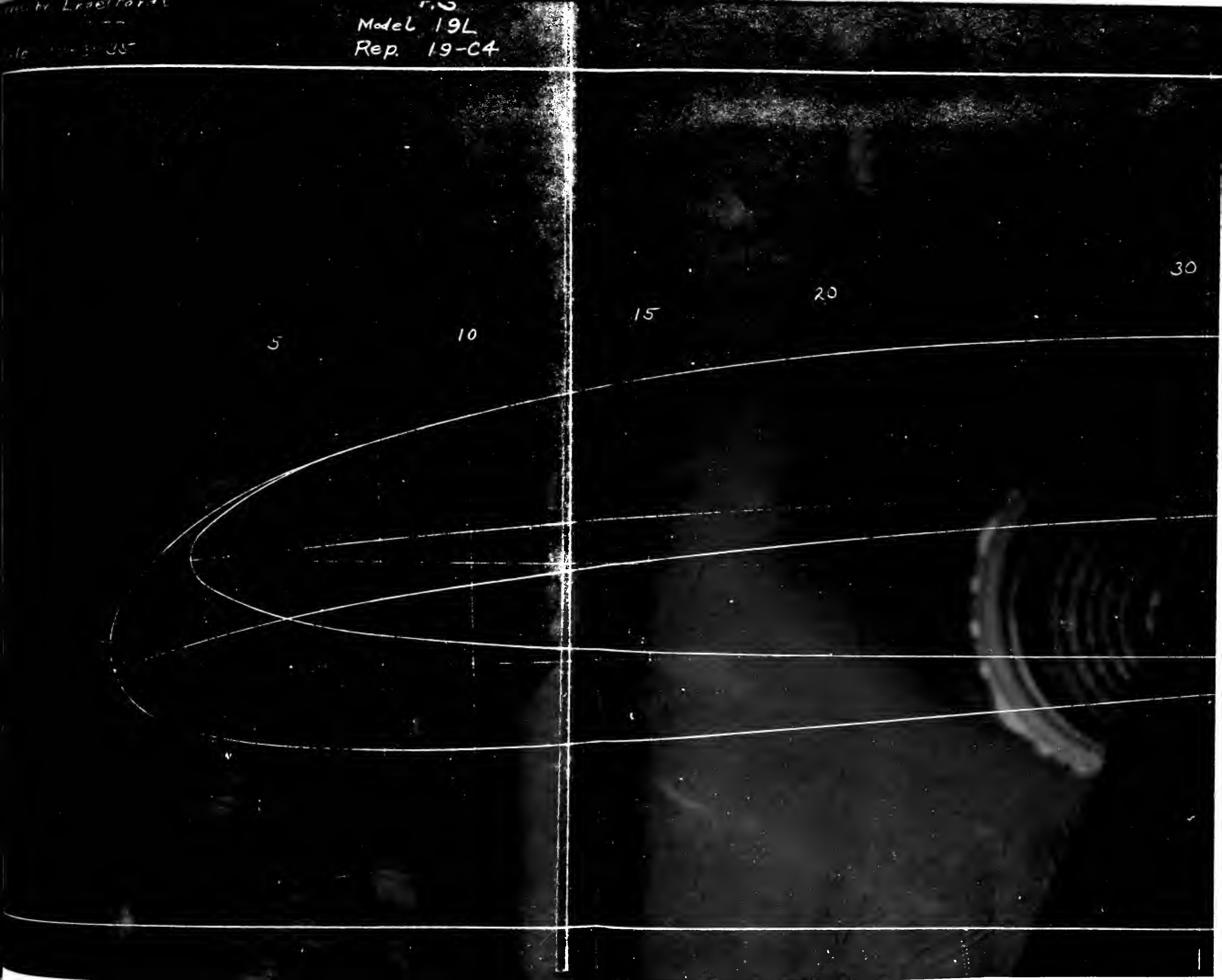
...gelhardt

...9-35

CURTISS-WRIGHT AIRPLANE CO.
ROBERTSON, MO.

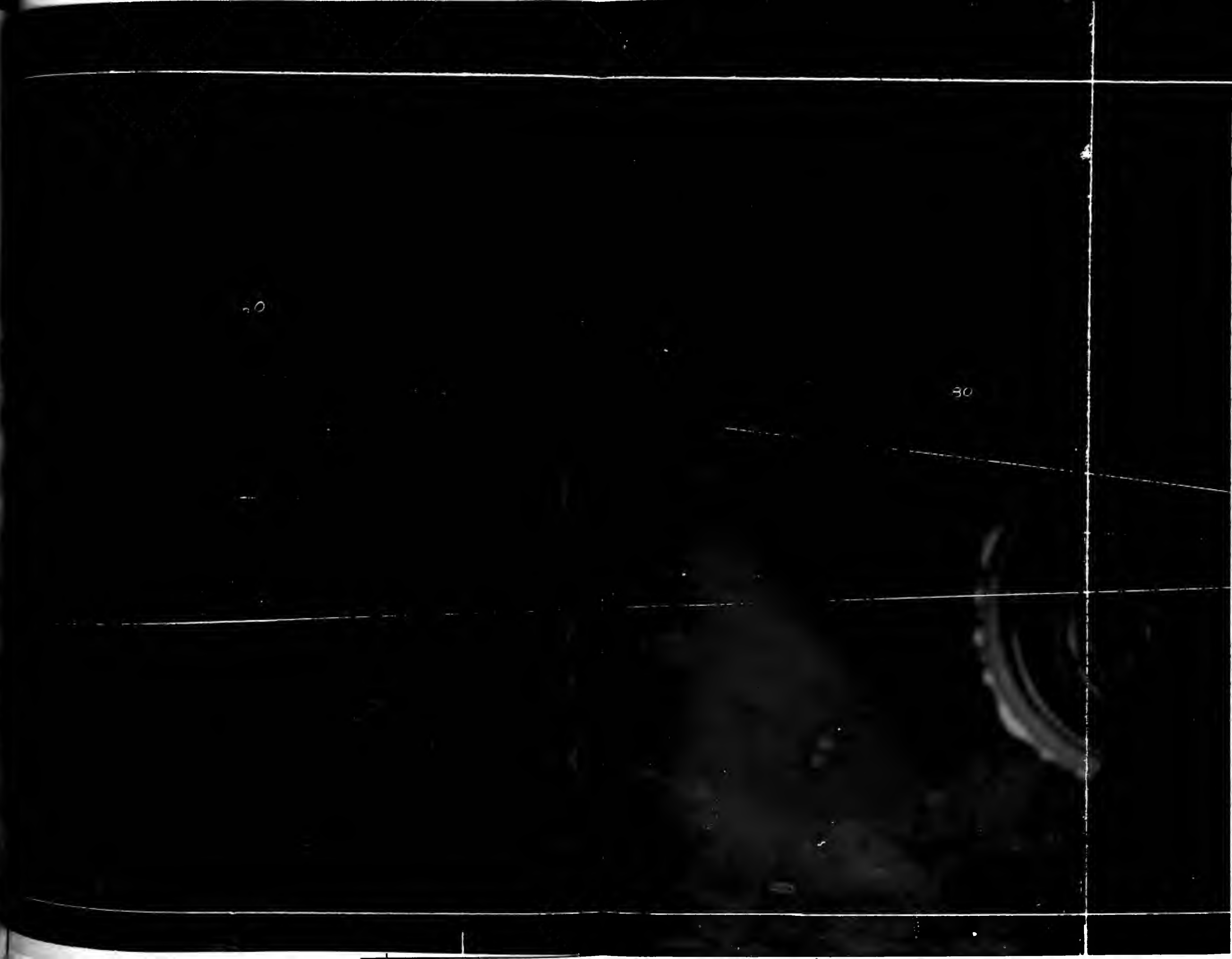
Page 2
Model 191
Report 19-C4


A
15''
351



Model 19L
Rep. 19-C4
5
10
15
20
30


40
35



P. 3

987

Defendants' Exhibit OOO—(Continued)

Report No. 19-Y3

Curtiss-Wright Airplane Co.

Robertson, Mo.

Engineering Department

Curtiss-Wright Sparrow, Model 19L
(2PCLM)
1 Lambert R-266, 90 H.P. Engine

Flight Tests
Sparrow No. 1

Submitted By

C. W. Scott. Section..........
No. of Pages...... Date 9/19/35
Revisions ....

Pages........
Date Affected Remarks

....

....

Flight No. 2

Date, 7/29/35
Take-Off, 6:30 A.M.
Land, 7:15 A.M.
Place, Lambert Field
Sta. Temp. 92.
Sta. Wind, E. 5
Sta. Bar., 29.98
Gross Weight, 1,668
C. G. 22.65
Propeller Diameter, 6.5′
Propeller Pitch, 16 Deg.

Pilot, E. K. Campbell
Observer, C. W. Scott

Defendants' Exhibit OOO—(Continued)

Alt.	Temp.	I.A.S.	RPM	Oil
2000	83	94	2060	154
True C.A.S.		97	Speed	103

Eng. Compt.	Thermocouples					Man. Rdgs	(″$H^2 0$)
	#1	#2	#3	#4	#5		
159	530	530	390	380	460	Ex. #1	8½
						Ex. #2	4
						Nose	7.2
						Eng.	6.5
						Ram	10.0

Fuel Pressure, 2.5
Oil Pressure, 65.

Full Throttle.				CAS.	TAS
2000	80	105	2270	109	115.3

Power Stall, Flaps Up, 54 I.A.S.,
Power Off, Flaps Down, 52 ″

47 C.A.S. Vicious Stall—Fell to right.

44 C.A.S. Vicious Stall—Fell to right.

On landing, it was found the L.H. landing gear did not stay up.

After this flight the following work was done on the ship:

1. Repair safety catch, left hand gear.
2. Install strings on right hand wing.

The takeoff characteristics were poor, no doubt due in part to the fact that the fixed slot was stalled thruout take-off. The time to accelerate to flying speed seems too much. The take-off speed was approximately 50 mph indicated air speed. Landing speed, 45 mph.

None of these speeds are calibrated.

Defendants' Exhibit OOO—(Continued)

On this flight considerable back pressure was observed on #1 and #2 cylinders. These two cylinders were also running at the highest head temperature. A new manifold will be tried with larger tubing on these two cylinders to correct this trouble.

Previous to this flight the ship was weighed and the C.G. located at 22.65% of the M.A.C.

Results of this flight:

1. Ailerons good but too much lag.
2. Elevators very sensitive.
3. Rudder light and mushy at stall.
4. Stalls vicious and sudden with no warning.

The following work was done for flight #3:

1. Remove and inspect safety catch on L.H. landing gear.
2. Install silk threads on R.H. wing.

Report No. 19-Y4

Curtiss-Wright Airplane Co.

Robertson, Mo.

Engineering Department

Curtiss-Wright Coupe, Model 19L
1 Lambert R-266, 90 H.P. Engine

Summary of Flight Tests
Coupe No. 1

Submitted By

C. W. Scott Section

No. of Pages 15 Date Nov. 22, 1935.

Defendants' Exhibit OOO—(Continued)

Revisions

Pages———

Date Affected

Remarks

. . . .

. . . .

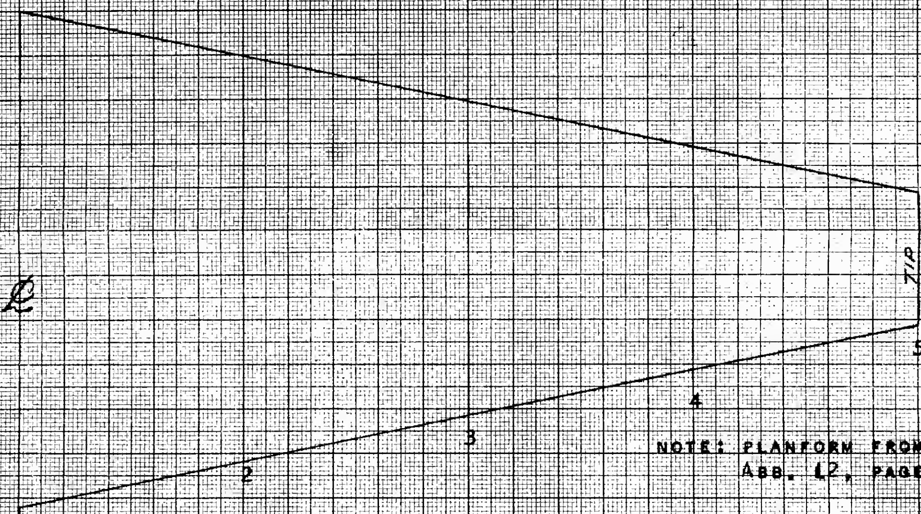
Introduction

The Model 19L has been run through a long series of very special flight tests. The purpose of this report is to draw conclusions from the results obtained rather than to go in to the detail of all of the different phases of the flight tests conducted. Briefly, the purpose of all of these tests has been first, to eliminate what was considered an undesirable stall characteristic of the basic airplane; second, to obtain satisfactory power plant cooling and operation; and third, a thorough check on the aerodynamic characteristics of the ship.

The results of the flying that has been accomplished on the Model 19L as finally modified are contained briefly in this report.

Admitted November 24, 1950.

Case No. 10930-Y
CARBEN vs. CONSOLIDATED
DFT EXHIBIT RRR
Date No. IDENTIFICATION
Date NOV 24 1950 No. RRR IN EVIDENCE
Clerk, U. S. District Court, Sou. Dist. of Calif. "LUFTFAHRTFORSCHUNG" (1938) pp. 604-613
Deputy Clerk (REFER EXHIBIT #22)

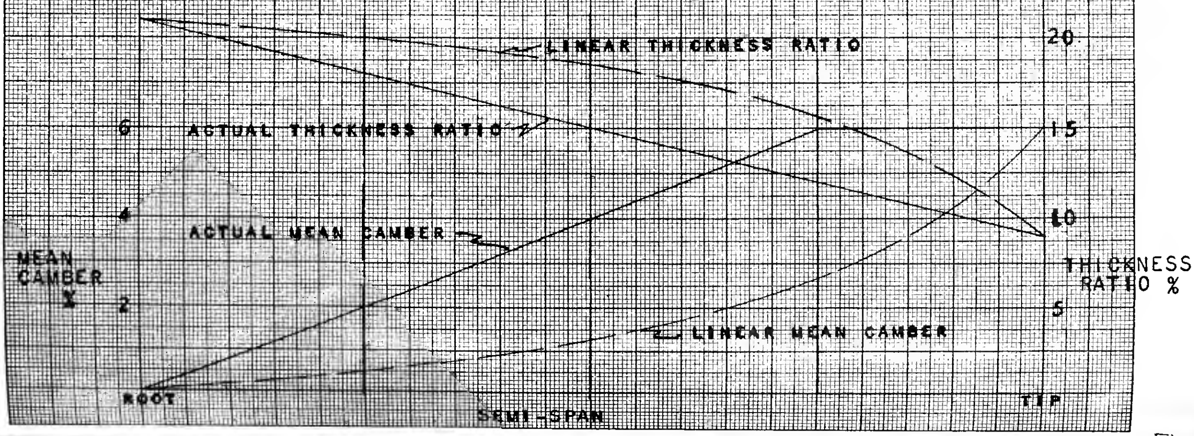


NOTE: PLANFORM FROM ABB. 12, PAGE 612

SECTION	1	2	3	4	5
LOCATION % SEMI-SPAN	0	25	50	75	100
MEAN CAMBER % OF CHORD	0	2	4	6	6
THICKNESS RATIO % OF CHORD	21	18	15	12	9

LINEAR DISTRIBUTION

MEAN CAMBER % OF CHORD	0	0.5	1.3	2.7	6
THICKNESS RATIO % OF CHORD	21	20	18.5	15.6	9



DEFENDANTS' EXHIBIT 000

SECTION DERIVED FROM FIG. 4

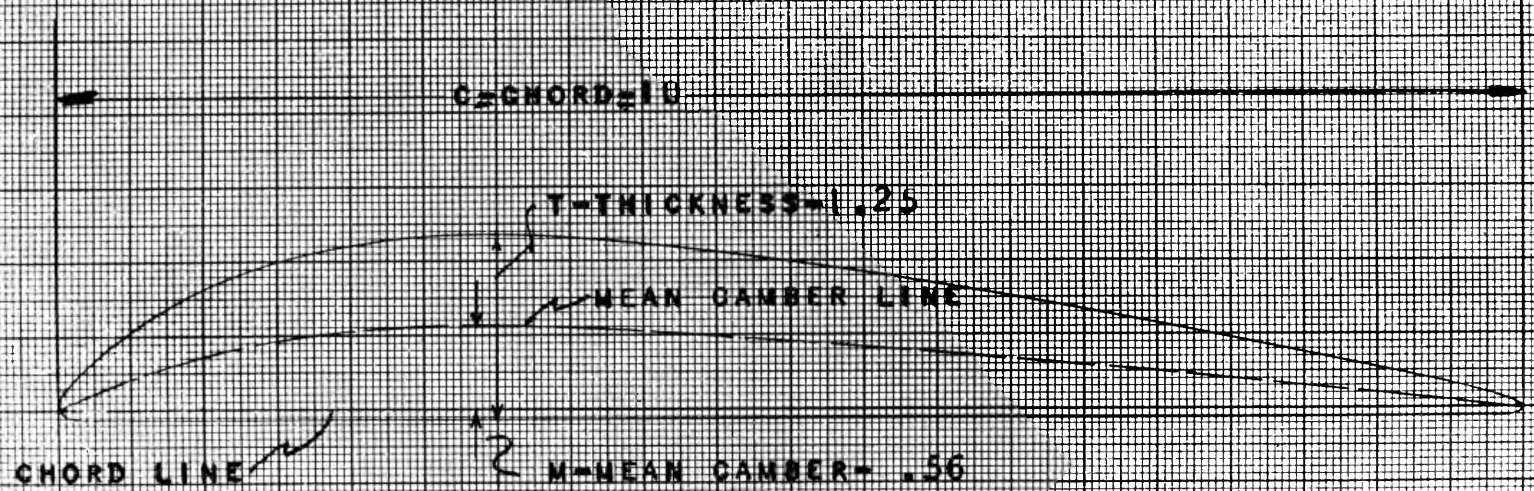


FIGURE 4

U.S. PATENT #1,547,644 - CRONSTEDT

MEAN CAMBER = M/C = 5.6% (OF CHORD)

THICKNESS RATIO = T/C = 12.5% (OF CHORD)

P.

SECTION DERIVED FROM FIG. 5

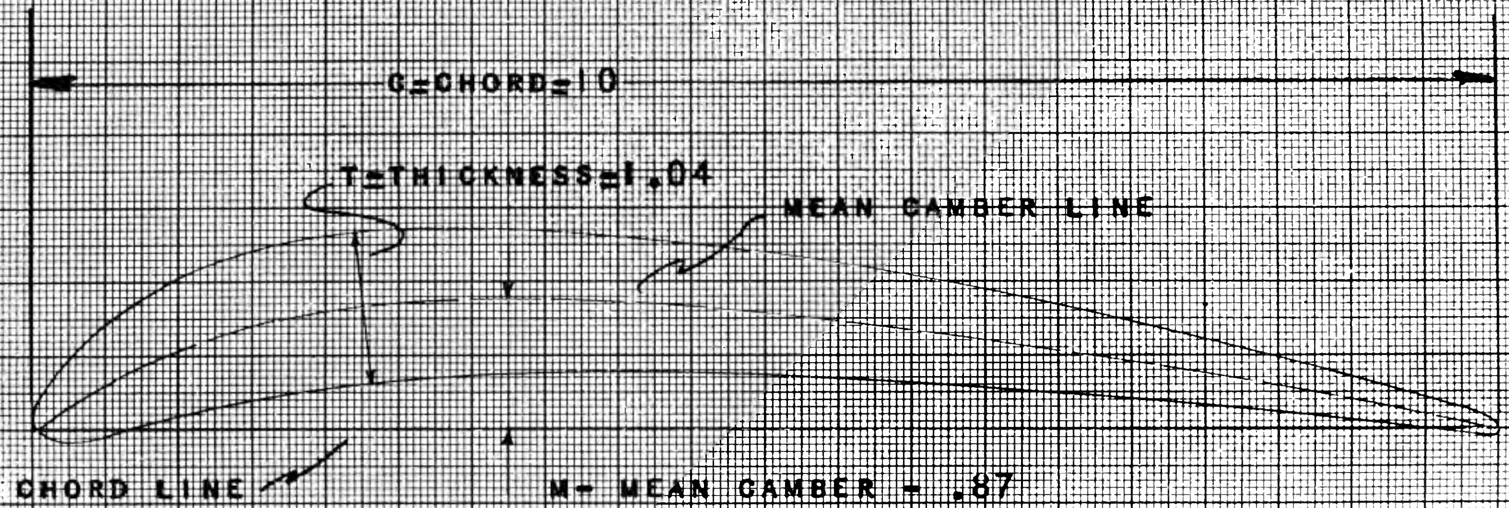


FIGURE 5

U.S. PATENT #1,547,644 - CRONSTEDT

MEAN CAMBER = M/C = 8.7% (OF CHORD)

THICKNESS RATIO = T/C = 10.4% (OF CHORD)

SECTION DERIVED FROM FIG. 3

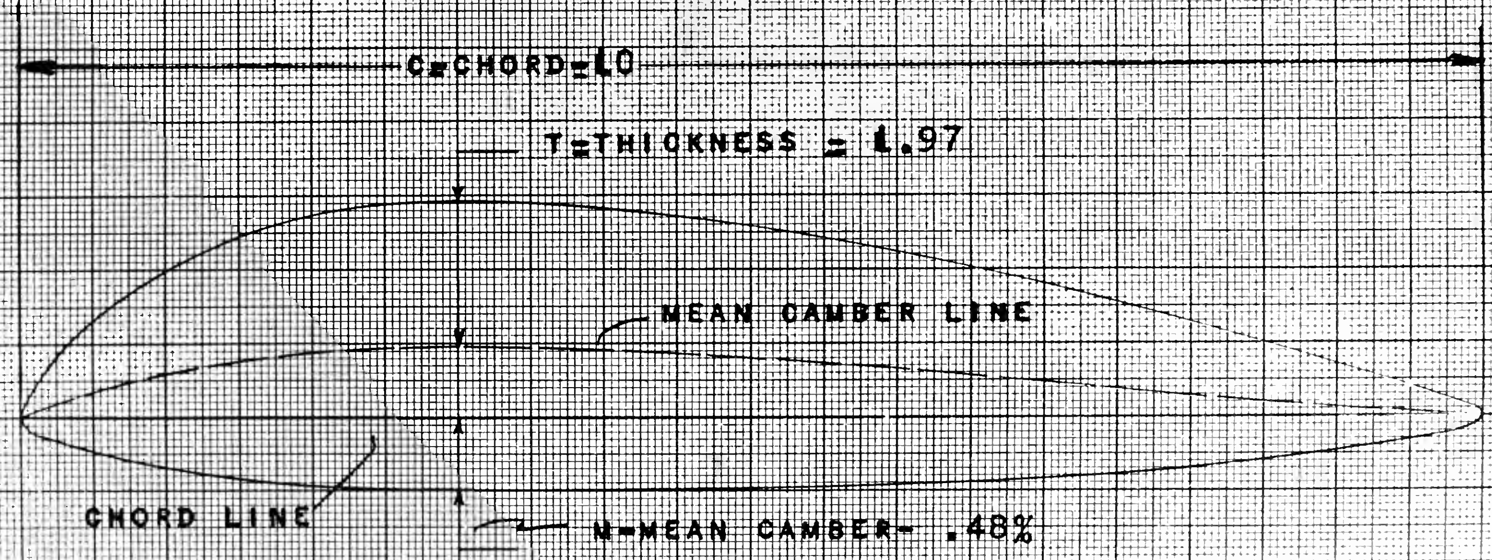


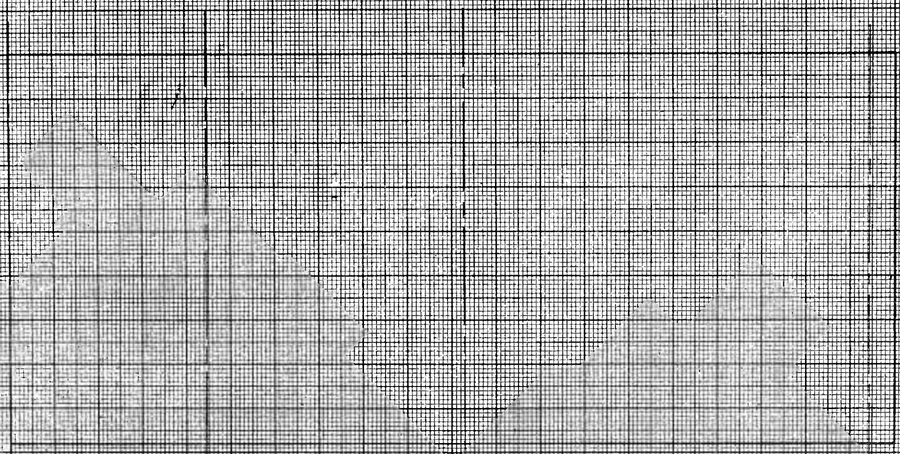
FIGURE 3

US. PATENT #1,547,644 - CRONSTEDT

MEAN CAMBER = M/C = 4.8% (OF CHORD)

THICKNESS RATIO = T/C = 19.7% (OF CHORD)

U.S. PATENT #1,547,644 - CRONSTEDT



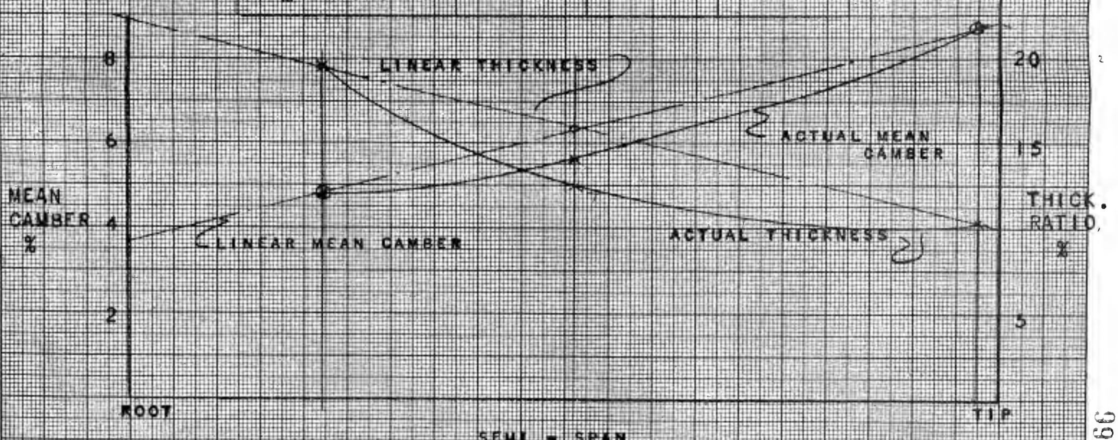
1 (FIG. 3) 2 (FIG. 4) 3 (FIG. 5)

NOTE: PLANFORM DERIVED FROM FIGS. 1 AND 2 OF PATENT

SECTION	1	2	3
LOCATION % OF SEMISPAN	22.5	51.5	97.0
MEAN CAMBER % OF CHORD	4.8	5.6	8.7
THICKNESS RATIO % OF CHORD	19.7	12.5	10.4

LINEAR DISTRIBUTION

MEAN CAMBER % OF CHORD	4.8	6.3	8.7
THICKNESS RATIO % OF CHORD	19.7	16.2	10.4



Admitted November 24, 1950.

997

Garbell vs Consol Vultee et al

Defts EXHIBIT 222

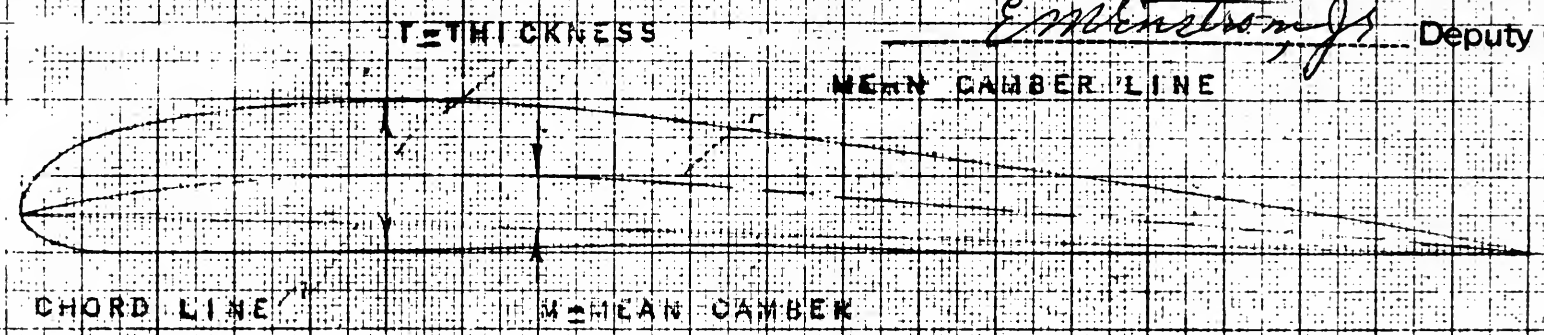
Date ______ No. ______ IDENTIFICATION

Date NOV 27 1950 No. 222 IN EVIDENCE

Clerk, U. S. District Court, Sou. Dist. of Calif.

E. M. Enstrom, Jr. Deputy Clerk

C = CHORD



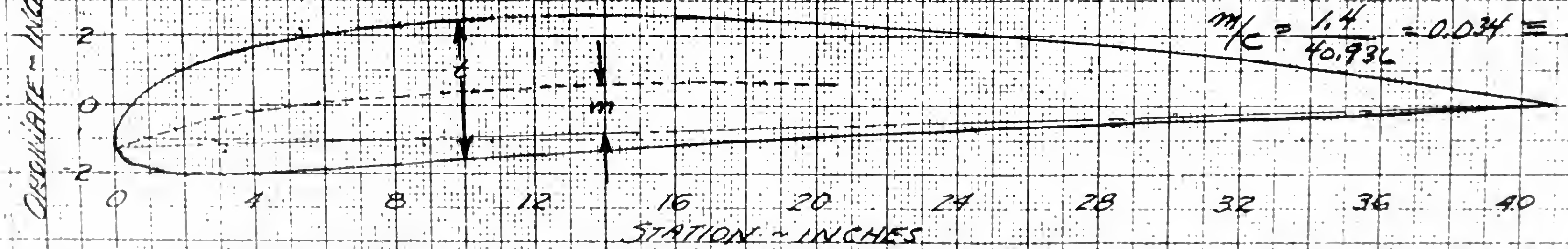
C W 19

MEAN CAMBER = M/C = 3.4% (OF CHORD)

THICKNESS RATIO = T/C = 10% (OF CHORD)

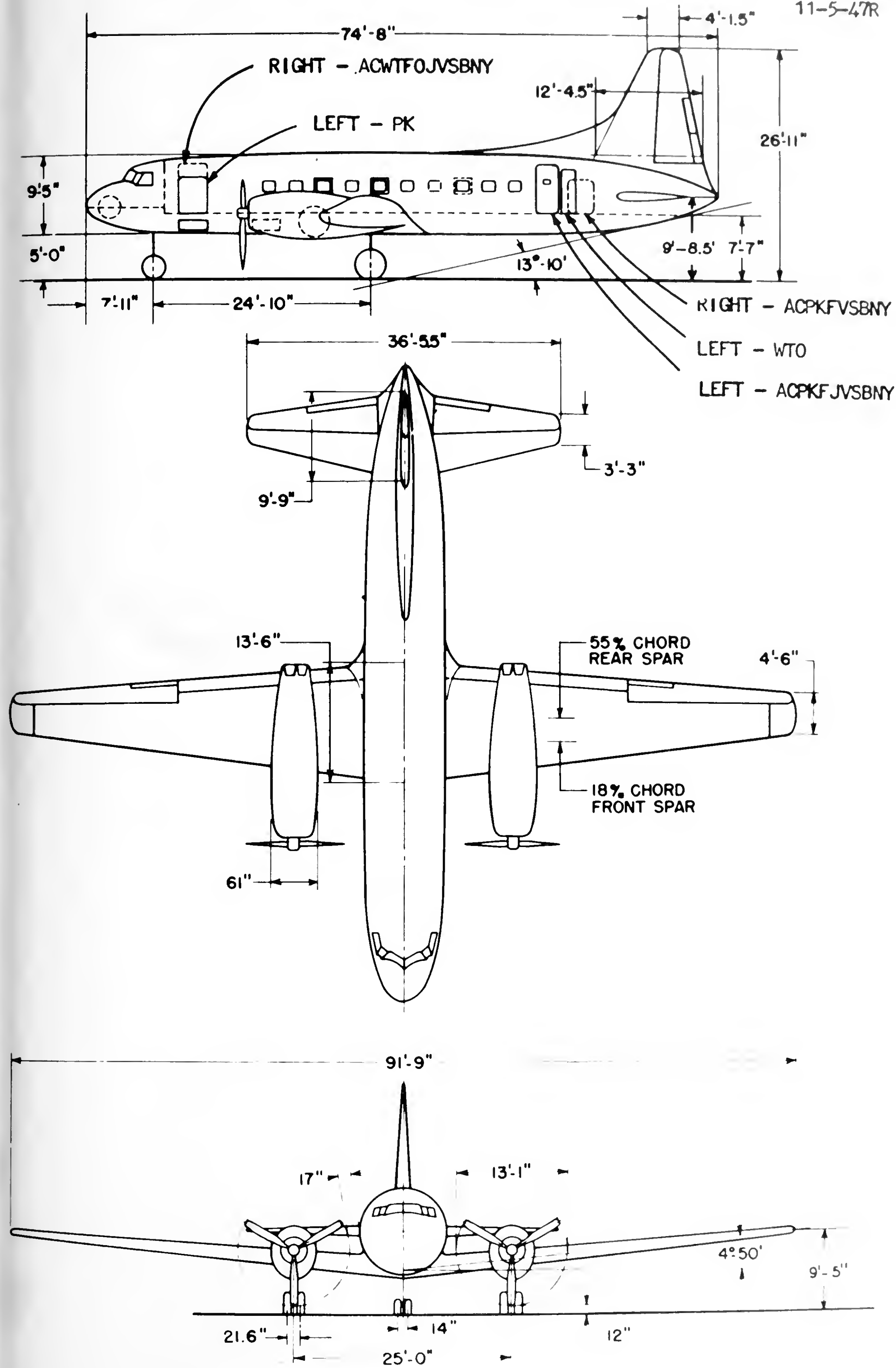
$t/c = \frac{4.10}{40.936} = 0.100 = 10.0\% c$

$m/c = \frac{1.4}{40.936} = 0.034 = 3.4\% c$



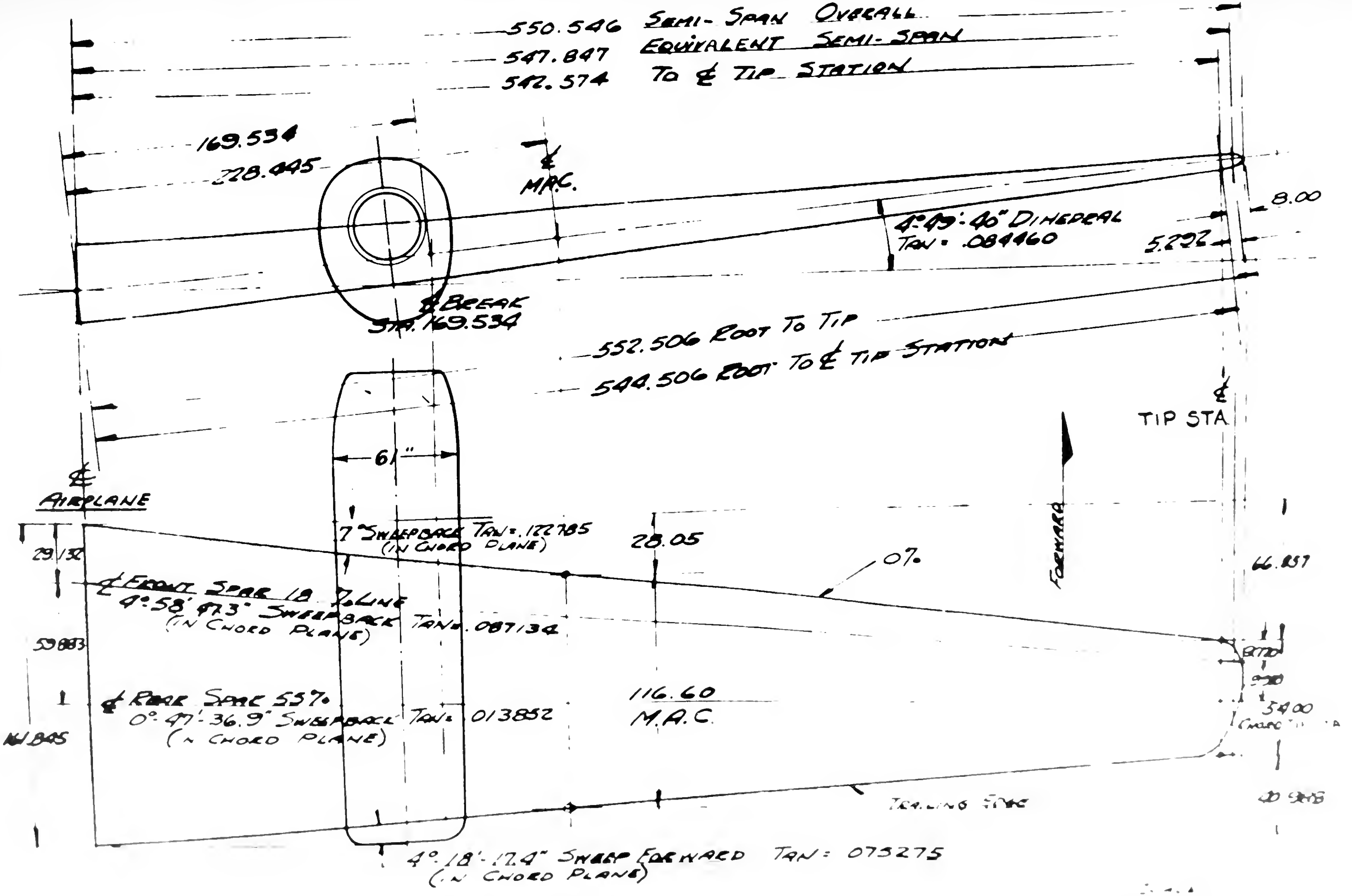
SECTION DERIVED FROM ORDINATES GIVEN ON CURTISS-WRIGHT DWG. NO. 19-03-220

DEFENDANTS' EXHIBITS REFERRED TO IN ANSWER TO INTERROGATORY XVII

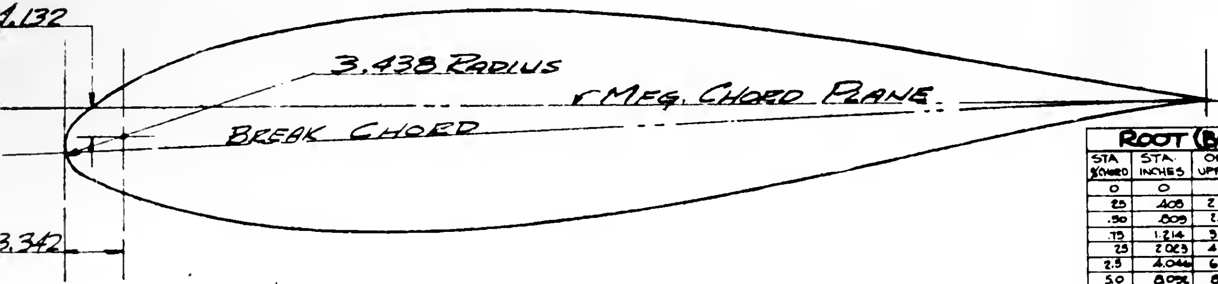

74'-8"
4'-1.5"
RIGHT - ACWTFOJVSBNY
LEFT - PK
12'-4.5"
26'-11"
9'-5"
5'-0"
9'-8.5' 7'-7"
13°-10'
7'-11"
24'-10"
RIGHT - ACPKFVSBNY
LEFT - WTO
LEFT - ACPKFJVSBNY
36'-5.5"
3'-3"
9'-9"
13'-6"
55% CHORD
REAR SPAR
4'-6"
18% CHORD
FRONT SPAR
61"
91'-9"
17"
13'-1"
4°50'
9'-5"
21.6"
14"
12"
25'-0"

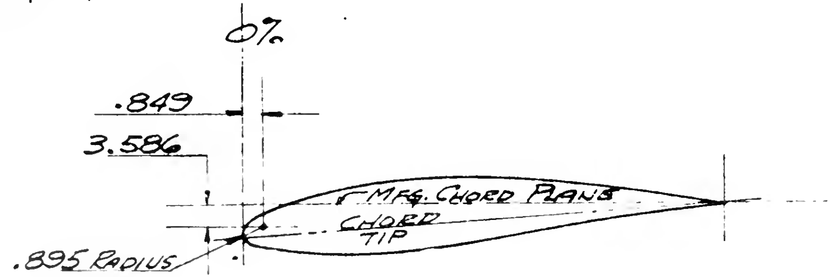
THREE VIEW

GENERAL WING DIMENSIONS



EXHIBIT





"ALL ORDINATES GIVEN FROM MFG. CHORD PLANE"

N.A.C.A. 63,4-120

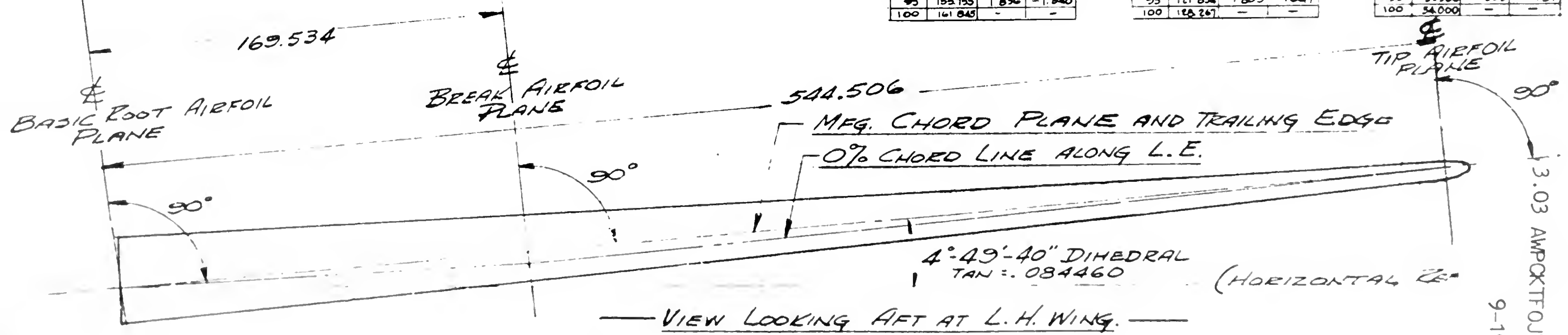
ROOT (BASIC)			
STA %CHORD	STA. INCHES	ORD UPPER	ORD LOWER
0	0	—	0
.25	.406	2.232	-1.790
.50	.809	2.997	-2.367
.75	1.214	3.600	-3.166
1.25	2.023	4.500	-4.048
2.5	4.044	6.102	-5.388
5.0	8.092	8.307	-7.670
7.5	12.138	10.066	-9.244
10	16.185	11.488	-10.820
15	24.277	13.680	-12.512
18	29.152	14.681	-13.402
20	32.368	15.243	-13.900
25	40.461	16.323	-14.844
30	48.554	16.839	-15.320
35	56.646	16.982	-15.320
40	64.738	16.660	-14.947
45	72.830	16.040	-14.270
50	80.923	15.166	-13.360
55	89.015	14.066	-12.288
60	97.107	12.768	-11.016
61.9999	100.334		
65	105.198	11.305	-9.625
70	113.292	9.787	-8.188
75	121.384	8.244	-6.774
80	129.476	6.689	-5.382
85	137.568	5.113	-4.000
90	145.661	3.506	-2.648
95	153.753	1.836	-1.340
100	161.845	-	-

N.A.C.A. 63,4-419

BREAK (169.534)			
STA. %CHORD	STA. INCHES	ORD. UPPER	ORD. LOWER
0	0	—	-4.537
.25	.521	-2.481	-5.796
.50	.641	-1.941	-6.319
.75	.962	-1.306	-6.741
1.25	1.603	-.793	-7.556
2.5	3.207	+.557	-8.582
5.0	6.415	2.366	-9.650
7.5	9.620	4.161	-10.519
10	12.827	5.302	-11.205
15	19.240	7.648	-12.189
18	23.088	8.663	-12.592
20	25.655	9.253	-12.801
25	32.067	10.431	-13.099
30	38.480	11.281	-13.097
35	44.893	11.608	-12.793
40	51.307	11.670	-12.157
45	57.720	11.477	-11.361
50	64.133	11.084	-10.420
55	70.547	10.488	-9.365
60	76.960	8.690	-8.205
61.9999	79.518		
65	83.374	8.730	-4.974
70	89.787	7.762	-5.751
75	96.200	6.109	-4.585
80	102.614	5.600	-3.481
85	109.027	4.484	-2.441
90	115.440	3.179	-1.480
95	121.854	1.809	-.647
100	128.267	—	—

N.A.C.A. 63,4-515

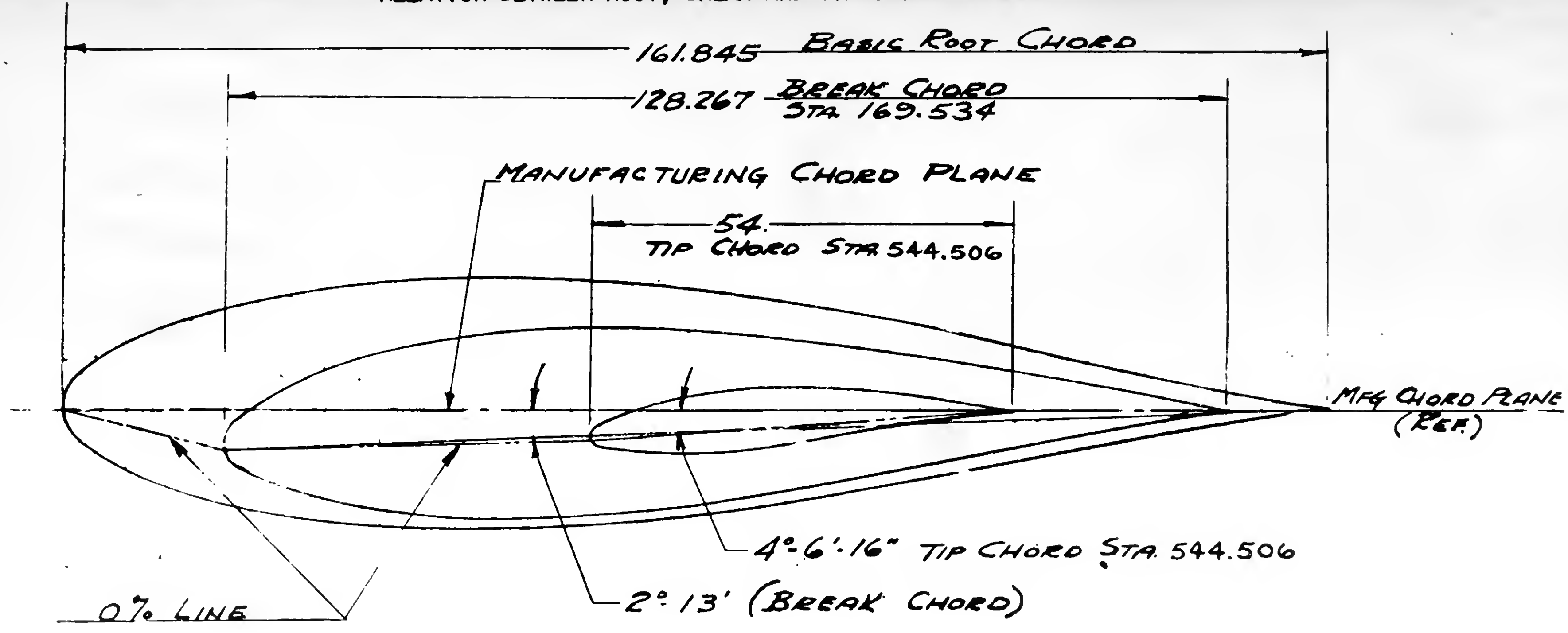
TIP (544.506)			
STA. %CHORD	STA INCHES	ORD. UPPER	ORD. LOWER
0	-	-	-3.875
.25	.135	-3.000	-4.135
.50	.270	-2.800	-4.300
.75	.405	-2.640	-4.417
1.25	.675	-2.510	-4.600
2.5	1.350	-1.855	-4.875
5.0	2.700	-1.060	-5.170
7.5	4.050	-.480	-5.353
10	5.400	+.125	-5.477
15	8.100	1.028	-5.625
18	9.720	1.469	-5.649
20	10.800	1.755	-5.652
25	13.500	2.285	-5.598
30	16.200	2.682	-5.445
35	18.900	2.943	-5.193
40	21.600	3.085	-4.865
45	24.300	3.145	-4.475
50	27.000	3.115	-4.045
55	29.700	3.017	-3.587
60	32.400	2.852	-3.095
61.9999	33.477	—	
65	35.100	2.639	-2.595
70	37.800	2.385	-2.107
75	40.500	2.100	-1.640
80	43.200	1.790	-1.200
85	45.900	1.460	-.805
90	48.600	1.092	-.465
95	51.300	.669	-.187
100	54.000	—	—



— VIEW LOOKING AFT AT L.H. WING. —

AWPCKTFOJVSBNY
9-19-46R

RELATION BETWEEN ROOT, BREAK AND TIP CHORD PLANES



VIEW SHOWING RELATION BETWEEN ROOT, BREAK, AND TIP CHORD PLANES AND MANNER IN WHICH THE DEGREE OF WASHOUT IN WING IS MEASURED.

INCIDENCE ANGLE 4° REF.

3.0201 AWPCKTFOJVSBNY
9-19-46R

EXHIBIT 4

Consolidated Vultee Aircraft Corporation
San Diego Division

Page 10
Report No. ZD-240-040
Model 240
Date

3.0 Characteristics

3.5 Dimensions and Areas:

3.5.1 Wing Group:

Airfoil Section Designation:

Root .... NACA 63.4—120
30.7% Semispan .... NACA 63.4—419
Tip .... NACA 63.4—515
Aerodynamic Washout .... 1° 12′
Wing Area .... 817 sq. ft.
Span (overall) .... 91 ft. 9 in.
Root Chord .... 13 ft. 6 in.
Tip Chord .... 4 ft. 6 in.
Taper Ratio (approximate) 3:1
Incidence Root .... 4°
Dihedral (reference Plane) .... 4° 50′
Sweepback (at 40% chord) .... 2° 30′
Aspect Ratio 10
Mean Aerodynamic Chord (true) ... 9 ft. 8.6 in.

3.5.3 Body Group:

Maximum Fuselage Cross Section:

Height .... 9 ft. 5 in.
Width .... 9 ft. 5 in.

Length, overall .... 74 ft. 8 in.
Height over tail (3-point position) .. 26 ft. 11 in.
Thread of main wheel .... 25 ft. 0 in.

No. 12885

United States
Court of Appeals
for the Ninth Circuit.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION and AMERICAN AIR LINES, INC.,

Appellants,

vs.

MAURICE A. GARBELL, INC., and GARBELL RESEARCH FOUNDATION,

Appellees.

SUPPLEMENTAL
Transcript of Record

Volume V
Book of Exhibits
(Pages 1007 to 1137)

Appeal from the United States District Court for the Southern District of California, Central Division.

Phillips & Van Orden Co., 870 Brannan Street, San Francisco, Calif.

INDEX

[Clerk's Note: When deemed likely to be of an important nature, errors or doubtful matters appearing in the original certified record are printed literally in italic; and, likewise, cancelled matter appearing in the original certified record is printed and cancelled herein accordingly. When possible, an omission from the text is indicated by printing in italic the two words between which the omission seems to occur.]

PAGE

Exhibits, Defendants':

Ex. A—Report on Airfoil Selection for the Revised Two-Engine Tailless Design ZA-101.....................1007

EE—Glenn L. Martin Co. Engineering Report No. 1326...................1071

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DEFENDANTS' EXHIBIT A

Page 1 of 60

Consolidated Aircraft Corporation
San Diego, California

Model..........Airplane Report No. ZA-101

Report on Airfoil Selection for the Revised
Two-Engine Tailless Design
ZA-101

February 25, 1944

By Abraham Firel
M. A. Garbell
M. Rogers

Approved: [Illegible]

Page 2 of 60

Consolidated Vultee Aircraft Corporation
San Diego Division

Model..........Airplane Report No. ZA-101

Foreword

This report summarizes the concepts and procedures used in the selection of airfoils for the revised two-engine tailless design.

Defendants' Exhibit A—(Continued)

Page 3 of 60

Consolidated Vultee Aircraft Corporation
San Diego Division

Model..........Airplane Report No. ZA-101

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Appendix D—Leading Edge Radius............56

Defendants' Exhibit A—(Continued)

Consolidated Vultee Aircraft Corporation
San Diego Division

Model..........Airplane Report No. ZA-101

Summary

New airfoils were selected for the revised two-engine tailless design to satisfy the following design requirements:

1. Good stalling characteristics with elevators neutral and deflected upward;

2. More conservative chordwise load distribution to retard the premature separation observed on the original airfoils;

3. Higher maximum lift with flaps retracted;

4. Slightly greater positive pitching moment coefficient at zero lift to raise the trim lift coefficient.

The new airfoils selected are defined by the following parameters:

Airfoil Family	Design Lift Coefficient C_{li}	Maximum Thickness % Chord	Chordwise Load Parameters a	b	Spanwise Location
63,4	.1	22	.1	.59	Root
63,4	.3	18	.1	.59	.48 Semi-Span
63,4	.5	16	.1	.59	Tip

The three airfoils are to be placed in a tapered wing of aspect ratio 12, taper ratio 4:1, leading-edge sweepback 11°-24′, with one degree aerodynamic washout at .48 semi-span and at the tip, referred to the root chord.

Defendants' Exhibit A—(Continued)

Data used in the selection of these airfoils are given in the text of the report and in the appedices. The geometric characteristics of the airfols and wing may be obtained from the various tables and charts.

Consolidated Vultee Aircraft Corporation
San Diego Division

Model..........Airplane Report No. ZA-101

Airfoil Selection & Wing Design

Structural and balance considerations, rather than the aerodynamicist's judgment, often determine the design of conventional wings. However, as stability and good stalling characteristics were to be the major criteria in the design of the revised wing for the present tailless design, few of the usual restrictions were imposed on the aerodynamicist in the determination of wing airfoil section and wing twist.

Inasmuch as the original wing appeared to be subject to premature trailing-edge separation, the airfoil camber-line loading was modified to give as gentle a pressure recovery as possible and still have a specified positive pitching moment at zero lift The leading edge was, therefore, loaded more than was done on the original airfoil, and the load was then distributed more gradually along the chord.

Defendants' Exhibit A—(Continued)

The NACA 63,4-XXX family of airfoils was considered to be best for the present design.

Previous two-dimensional wind-tunnel tests of the original 63,4-221 (a=.27, b=.54) airfoil, proposed as the root section for a tailless design, indicated that there was a correlation factor of about 3 between theoretically calculated section-pitching moment coefficients and those obtained in the wind-tunnel. Examination of theoretical and experimental pressure distribution data indicated that the difference between theory and experiment was greatest near the rear portion of the airfoil.

Page 6 of 60

Consolidated Vultee Aircraft Corporation
San Diego Division

Model..........Airplane Report No. ZA-101

This difference between theoretical and experimental pressure distribution data has been examined by Robert M. Pinkerton (Reference 6). Pinkerton explains the difference as an effect of viscosity, which is neglected in the development of the theory. The viscosity of the air is observed as a frictional force producing drag on the airfoil. Since the layer of air that passes over the airfoil is slowed down by this frictional force, a low-energy boundary layer is produced. The boundary layer thickens towards the trailing edge of the airfoil. Since all pressures are transmitted normal to this boundary layer without

Defendants' Exhibit A—(Continued)

change (Pascal's Law), the actual pressure distribution measured over the airfoil is that existing over the contour formed by the boundary layer and not by the material airfoil. The differences between theory and experiment are, therefore, greatest over the aft portion of the airfoil where the boundary layer is thickest and the deviation from the true airfoil contour greatest.

The theoretical pitching-moments of the revised airfoils were, therefore, selected to give one-third the value needed to produce the required wing pitching-moment. The full-scale wing-alone pitching-moment coefficient at zero lift C_{m_0} for proper trim and stability was estimated to be 0.060. Theoretical airfoil pitching-moment coefficients at zero lift of 0.0065 at the root, 0.0195 at the 48% semi-span point, and 0.0325 at the tip were selected as proper values to give this required full-scale wing-alone moment.

Span-load distributions showed that twist distribution alone, as a means of obtaining satisfactory wing stalling characteristics, was isfactory from the viewpoint of drag. [A combination of camber ckness was selected to provide a desirable section maximum-lift-ient distribution and yet to maintain a high critical Mach number. g airfoil sections as selected to meet these criteria are:

3,4-422 (a = .1, b = .59)

i-span: 63,4-318 (a = .1, b = .59)

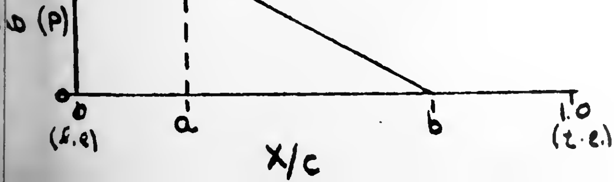
63,4-516 (a = .1, b = .59)

The aerodynamic washout required to obtain favorable stall eristics as well as reasonable drag values was found to be 1° root and splice, with no additional washout between the splice wing tip.]

AIRFOIL THEORY

The characteristic properties of a low-drag airfoil, i.e. the g-moment at zero lift, C_{mo}; the maximum lift-coefficient, $C_{\ell\,max}$; C_ℓ range and location of the minimum-drag region are determined nsiderable extent by the shape of the mean-camber-line, subject fication by the particular thickness distribution of the complete .

The mean-camber-line load distribution can be described by d parameters schematically represented below. The load is as constant from the leading edge to a station "a" on the chord airfoil and is assumed as linearly decreasing to zero from station station "b", the load remaining zero from station "b" to the g-edge of the airfoil.



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The formulae for the ordinates and slope of the camber line from Reference 7, are:

I. Ordinates:

$$y_c = \frac{C_{l_i}}{2\pi(a+b)}\left[\frac{1}{b-a}\left\{\frac{1}{2}(a-x)^2 \ln|a-x| - \frac{1}{2}(b-x)^2 \ln|b-x| + \frac{1}{4}(b-x)^2 - \frac{1}{4}(a-x)^2\right\} - x\ln x + g - hx\right] \qquad (1)$$

where:

$$g = -\frac{1}{b-a}\left[a^2\left\{\frac{1}{2}\ln a - \frac{1}{4}\right\} - b^2\left\{\frac{1}{2}\ln b - \frac{1}{4}\right\}\right]$$

$$h = \frac{1}{b-a}\left[\frac{1}{2}(1-a)^2\ln(1-a) - \frac{1}{2}(1-b)^2\ln(1-b) + \frac{1}{4}(1-b)^2 - \frac{1}{4}(1-a)^2\right] + g$$

II Slope

$$\frac{dy_c}{dx} = -\frac{C_{l_i}}{2\pi(a+b)}\left[\frac{1}{b-a}\left\{(a-x)\ln|a-x| - (b-x)\ln|b-x|\right\} + \left\{1 + \ln x + h\right\}\right] \qquad (2)$$

The design lift coefficient (C_{l_i})* which corresponds closely to [li]t-coefficient for lowest drag, (i.e. the lift coefficient located [th]e center of the low-drag range) is defined as

$$C_{l_i} = \int_0^1 \text{LOAD}\left[d\left(\frac{x}{c}\right)\right] = \int_0^1 P\left[d\left(\frac{x}{c}\right)\right]$$

[P] is the term called $_0P_b$ in Reference 1.

$$P = {}_0P_b = \frac{p - p_0}{\frac{1}{2}\rho V_0^2}$$

$$C_{l_i} = \int_0^a A\left[d\left(\frac{x}{c}\right)\right] + \int_a^b \frac{A\left(b-\frac{x}{c}\right)}{(b-a)}\left[d\left(\frac{x}{c}\right)\right] \qquad (3)$$

[(]A is the numerical value of the load P in the constant load range).

---

*The term C_{l_b} in Reference 1 is called C_{l_i} using E.N. Jacobs' [notati]on in the equation of the camber-line, since Jacobs' notation has [become] common in aeronautical use. C_{l_i} is here defined as the design [lift c]oefficient of the airfoil due to camber, whereas C_{l_b} is the actual lift coefficient of the airfoil and includes the effect of the thickness distribution.

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$$C_{l_i} = Aa + \frac{A}{(b-a)} b(b-a) - \frac{A}{2(b-a)}(b^2 - a^2)$$

$$= A(a+b) - \frac{A(b+a)}{2}$$

$$= \frac{A}{2}(a+b)$$

$$A = \frac{2C_{l_i}}{a+b} \qquad (4)$$

nt coefficient about the aerodynamic center of the airfoil is

$$C_{m_{\delta/c}} = \int_0^1 P(\delta/c - x/c)\, d\left(\frac{x}{c}\right) \qquad (5)$$

$$= \frac{2C_{l_i}}{a+b}\left\{ \delta/c\left[\int_0^a d(x/c) + \int_a^b \frac{b-\frac{x}{c}}{b-a} d\left(\frac{x}{c}\right)\right] - \int_0^a \frac{x}{c} d\left(\frac{x}{c}\right) - \int_a^b \frac{b-\frac{x}{c}}{b-a}\left(\frac{x}{c}\right) d\left(\frac{x}{c}\right)\right\}$$

$$= \frac{2C_{l_i}}{a+b}\left\{\frac{\delta}{c}\left[a + b\frac{(b-a)}{(b-a)} - \frac{1}{2}\frac{(b^2-a^2)}{(b-a)}\right] - \left[\frac{1}{2}a^2 + \frac{1}{2}\frac{b(b^2-a^2)}{b-a} - \frac{1}{3}\frac{(b^3-a^3)}{b-a}\right]\right\}$$

$$= \frac{2C_{l_i}}{a+b}\left\{\frac{\delta}{c}\left(\frac{a+b}{2}\right) - \frac{1}{6}(a^2 + ab + b^2)\right\} \qquad (5a)$$

is the aerodynamic center of the airfoil.

Figure 2 shows the relationship between C_{l_i}, C_{m_0}, "a", and "b" aerodynamic center at 0.265 . This chart can be used as a ary selection chart for "a" and "b" values to give a desired cal C_{m_0} for a given design lift coefficient C_{l_i}.

GENERAL THEORY OF AIRFOIL PRESSURE DISTRIBUTIONS

The pressure distributions for the airfoils treated in this were computed by the method outlined in References 1 and 2.

The principle involved is that of obtaining the pressure e velocity assuming a non-viscous, incompressible fluid in tional motion,

lli's equation is

$$H = p + \frac{1}{2}\rho V^2$$

Neglecting compressibility and viscosity the pressure at any of the upper surface is:

$$p_u = H - \frac{1}{2}\rho V_u^2$$

$= p_0 + \frac{1}{2}\rho V_0^2$ for some point in the free stream far from the l.

$$p_u - p_0 = \frac{1}{2}\rho\left[V_0^2 - V_u^2\right]$$

$$\frac{p_u - p_0}{\frac{1}{2}\rho V_0^2} = 1 - \left(\frac{V_u}{V_0}\right)^2$$

rly for the lower surface

$$\frac{p_l - p_0}{\frac{1}{2}\rho V_0^2} = 1 - \left(\frac{V_l}{V_0}\right)^2$$

ad on the airfoil is then

$$P = \frac{p_l - p_u}{\frac{1}{2}\rho V_0^2} = \left(\frac{V_u}{V_0}\right)^2 - \left(\frac{V_l}{V_0}\right)^2$$

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C_{l_b} is the basic lift due to the shape of the camber line and
ess distribution and C_{l_a} is the additional lift due to the change
le of attack*, the incremental load changes for airfoil of small
ess can be written.

$$\frac{V_u}{V_o} = \frac{V_s}{V_o} + \frac{\Delta u}{V_o} C_{l_b} + \frac{\Delta V_a}{V_o} C_{l_a}$$

$$\frac{V_l}{V_o} = \frac{V_s}{V_o} - \frac{\Delta u}{V_o} C_{l_b} - \frac{\Delta V_a}{V_o} C_{l_a}$$

$\frac{V_s}{V_o}$ is the velocity at any point on the symmetrical airfoil of the
hickness,

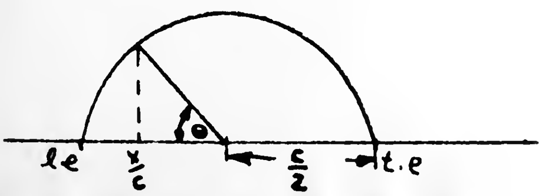
$\frac{\Delta u}{V_o} C_{l_b}$ is the velocity increment due to the camber line
g at C_{l_b}, and $\frac{\Delta V_a}{V_o} C_{l_a} = \frac{\Delta V_a}{V_o}(C_l - C_{l_b})$ is the velocity increment
symmetrical airfoil due to the change in angle of attack required
ain C_{l_a}. Substituting these values of velocity into the equations
e load

$$P = \left(\frac{V_u}{V_o}\right)^2 - \left(\frac{V_l}{V_o}\right)^2$$

It has been found (Reference 3) that the lift of airfoils of
thickness can be divided into two parts. One part is due only
shape of the airfoil camber line, and the other is due to the
of attack of the airfoil measured from an angle of attack, α_i,

$$\alpha_i = \frac{1}{\pi}\int_0^{\pi} \frac{dy_c}{d\left(\frac{x}{c}\right)} d\theta$$

$= \frac{1}{2}(1 - \cos\theta)$ as graphically represented below:



$$\left(\frac{V_s}{V_o} + \frac{\Delta u}{V_o} C_{l_b} + \frac{\Delta V_a}{V_o} C_{l_a}\right)^2 - \left(\frac{V_s}{V_o} - \frac{\Delta u}{V_o} C_{l_b} - \frac{\Delta V_a}{V_o} C_{l_a}\right)^2$$

$$= \frac{4V_s}{V_o}\left[\frac{\Delta u}{V_o} C_{l_b} + \frac{\Delta V_a}{V_o} C_{l_a}\right]$$

re,

$$= \int_0^1 \frac{4V_s}{V_o}\left[\frac{\Delta u}{V_o} C_{l_b} + \frac{\Delta V_a}{V_o} C_{l_a}\right]\left[d\left(\frac{x}{c}\right)\right]$$

$$\int_0^1 \frac{4V_s}{V_o}\left[\frac{\Delta u}{V_o} C_{l_b} + \frac{\Delta V_a}{V_o} C_{l_a}\right]\left[\frac{\delta}{c} - \frac{x}{c}\right]\left[d\left(\frac{x}{c}\right)\right]$$

$_2 = 0$

$$= \int_0^1 \frac{4V_s}{V_o}\left[\frac{\Delta u}{V_o} C_{l_b} + \frac{\Delta V_a}{V_o} C_{l_a}\right]\left[d\left(\frac{x}{c}\right)\right]$$

$$m_o = -\int_0^1 \frac{4V_s}{V_o}\left[\frac{\Delta u}{V_o} C_{l_b} + \frac{\Delta V_a}{V_o} C_{l_a}\right]\left[\frac{x}{c}\right]\left[d\left(\frac{x}{c}\right)\right]$$

he following method is employed to obtain C_{m_o}:

C_{l_i} is obtained by integrating the slope of the camber-line

$$\left(\frac{\Delta u}{V_o} C_{l_i}\right)_{\theta=\theta_o} = -\frac{1}{2\pi}\int_0^{2\pi} \frac{dy_c}{d(x/c)} \cot(\theta-\theta_o)\, d\theta$$

his integral can be evaluated by a numerical method given in

ces 1 and 4 and outlined in Appendix A of this report.

$_b$ is obtained by integrating the load due to C_{l_i}.

$$C_{l_b} = \int_0^1 \frac{4V_s}{V_a}\left(\frac{\Delta u}{V_o} C_{l_i}\right)\left[d\left(\frac{x}{c}\right)\right]$$

aking the area between the pressure distribution curves $\frac{p_l - p_o}{1/2\rho V_o^2}$

$\frac{-p_o}{1/2\rho V_o^2}$

$_a$ is set equal to $-C_{l_b}$ so obtained in order to get $C_l = 0$

general C_{l_b} is not equal to C_{l_i}.)

is computed from the integral

$$C_{m_0} = -\int_0^1 \frac{4V_s}{V_0}\left[\frac{\Delta u}{V_0} C_{l_b} + \frac{\Delta V_a}{V_0} C_{l_a}\right]\left[\frac{x}{c}\right]\left[d\left(\frac{x}{c}\right)\right]$$

TABLE OF NOMENCLATURE AND DEFINITIONS

= Basic sectional lift coefficient. The basic lift depends on the airfoil camber line and the thickness distribution, and is independent of angle of attack.

= Additional sectional lift coefficient. The additional lift depends only on the angle of attack as measured from α_i

= Angle of attack at which the additional lift is zero.

= Sectional design lift coefficient at which the additional lift of airfoil is zero. This lift coefficient occurs very close to the center of the minimum drag region. $C_{l_i} = C_{l_b}$ for airfoils of infinitely small thickness.

= Camber line load parameters, expressed in percent of chord.

= Aerodynamic center about which pitching moment is taken

$_m$= Sectional pitching moment.

= Basic load; identical to $_oP_b$ in Reference 1.

= static pressure at a point of the airfoil contour

$_o$ = static pressure in free stream

V_o^2= dynamic pressure in free stream.

θ = the angle whose cosine is $2\frac{x}{c} - 1$ (expressed in radians)

$/c$ = any point on the chord of the airfoil

= ratio of incremental velocity on airfoil to free stream velocity.

ript

u denotes upper surface

l denotes lower surface

s, t denotes thickness

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SAN DIEGO DIVISION

MODEL AIRPLANE REPORT No ZA-101

$\frac{\Delta u}{V_0} C_{l_i}$ OR $\frac{\Delta u}{V_0} C_{l_b}$ is the velocity due to the camber camber line and thickness respectively.

$\frac{\Delta V_a}{V_0} C_{l_a}$ = velocity on the symmetrical airfoil due to the change in angle of attack required to obtain C_{l_a}.

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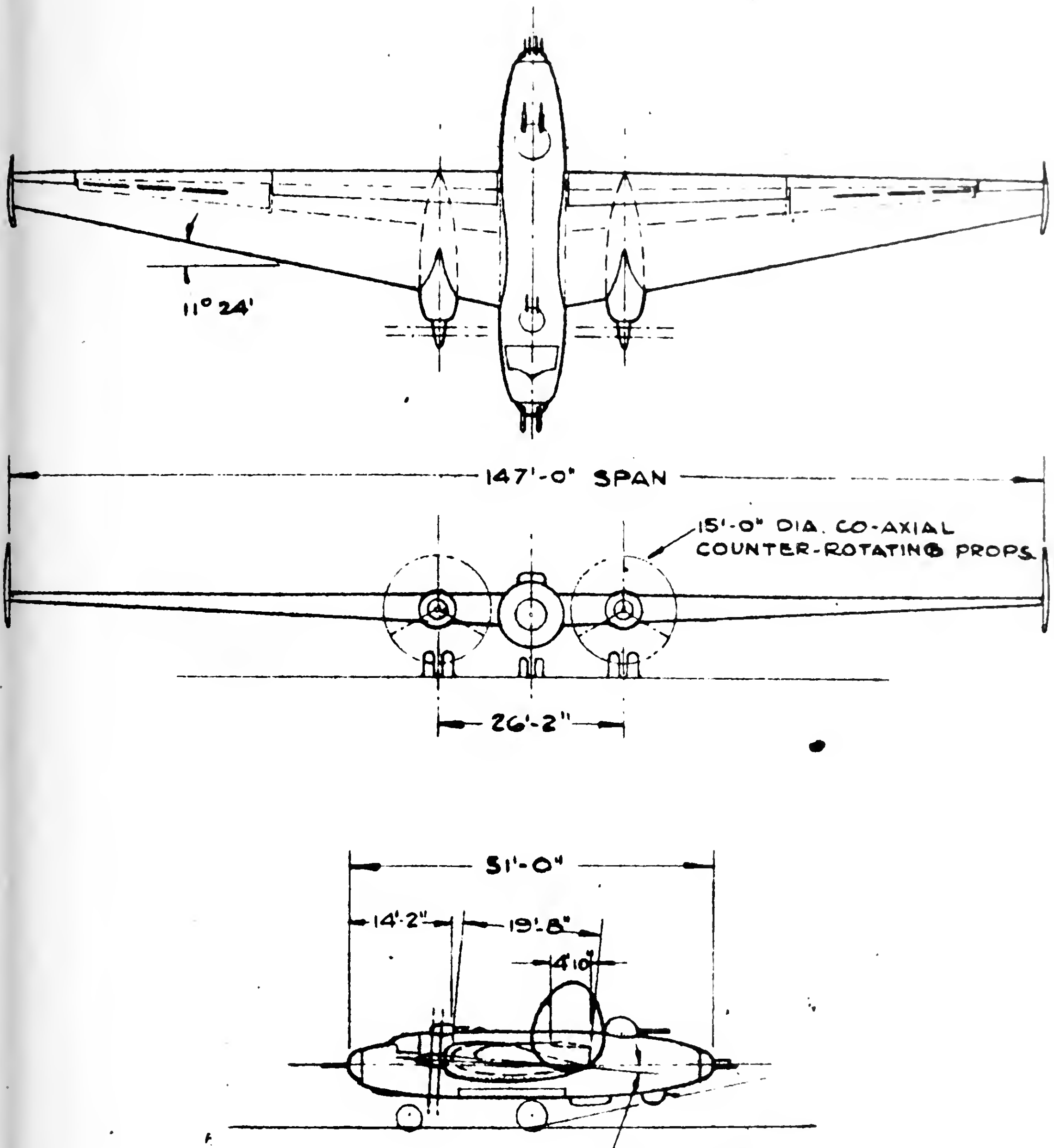
FIG 1



SCALE - 1/300

	NUÑEZ	2-8-44	GENERAL ARRANGEMENT REVISED 2-ENGINE TAILLESS DESIGN	PART NUMBER
D				
D			CONSOLIDATED AIRCRAFT CORPORATION LINDBERGH FIELD • SAN DIEGO, CALIFORNIA	
	MR	2-8-44		

FIG 2

PRELIMINARY CHART FOR THE ESTIMATION OF THEORETICAL SECTION PITCHING-MOMENT COEFF. (Q.C. @ .265 CHORD)

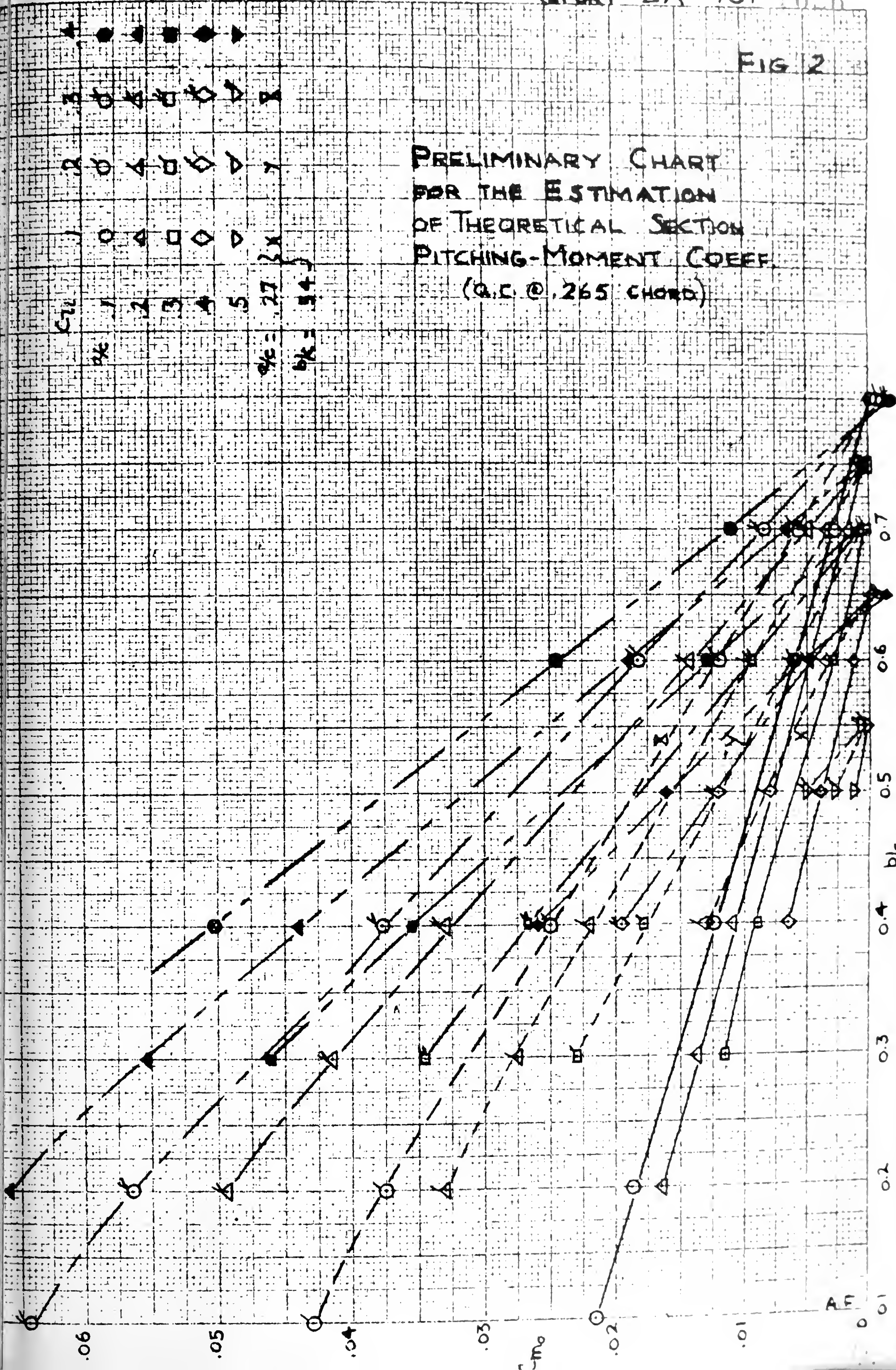
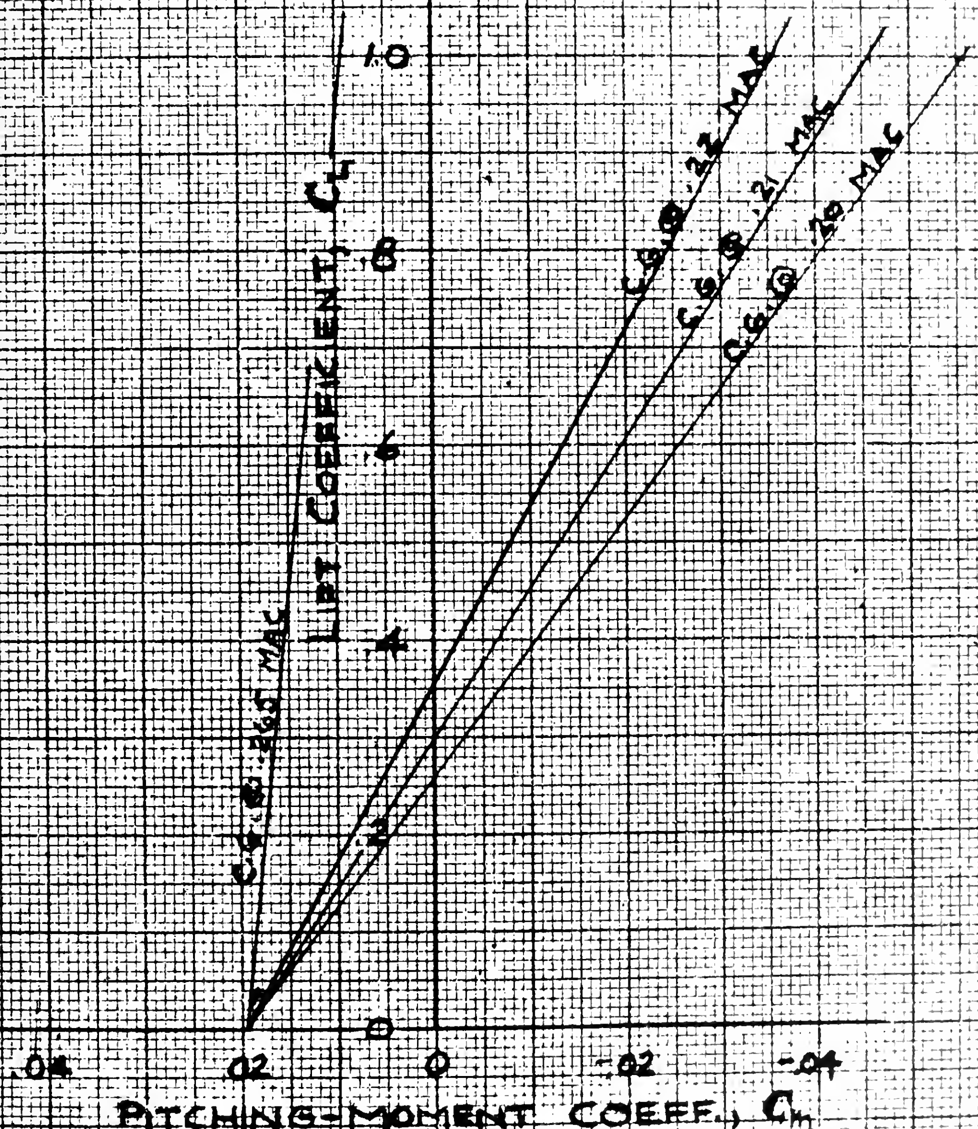
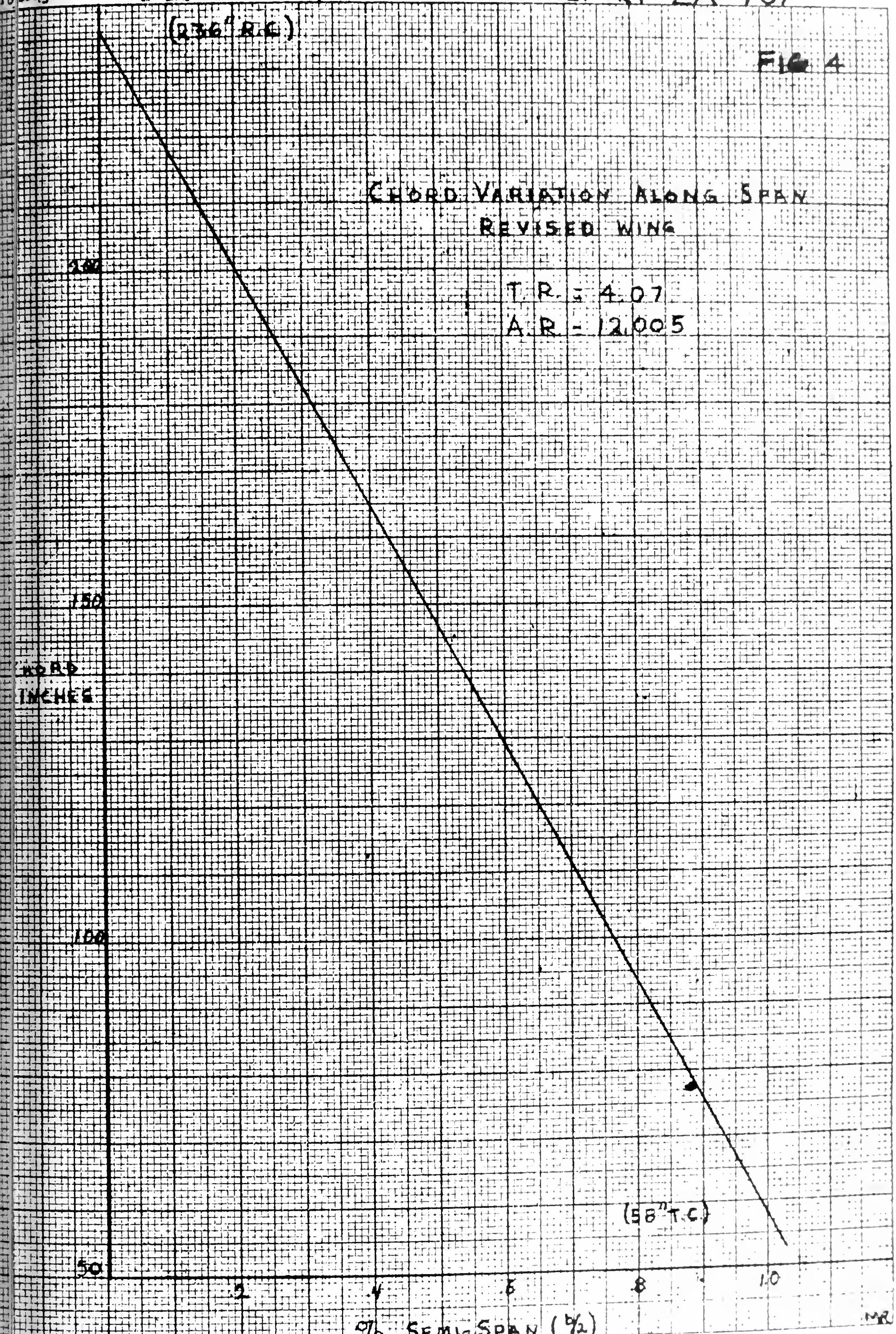


FIG. 3

REVISED WING
ESTIMATED PITCHING-MOMENT COEFF.
COMPLETE MODEL
(NO POWER)

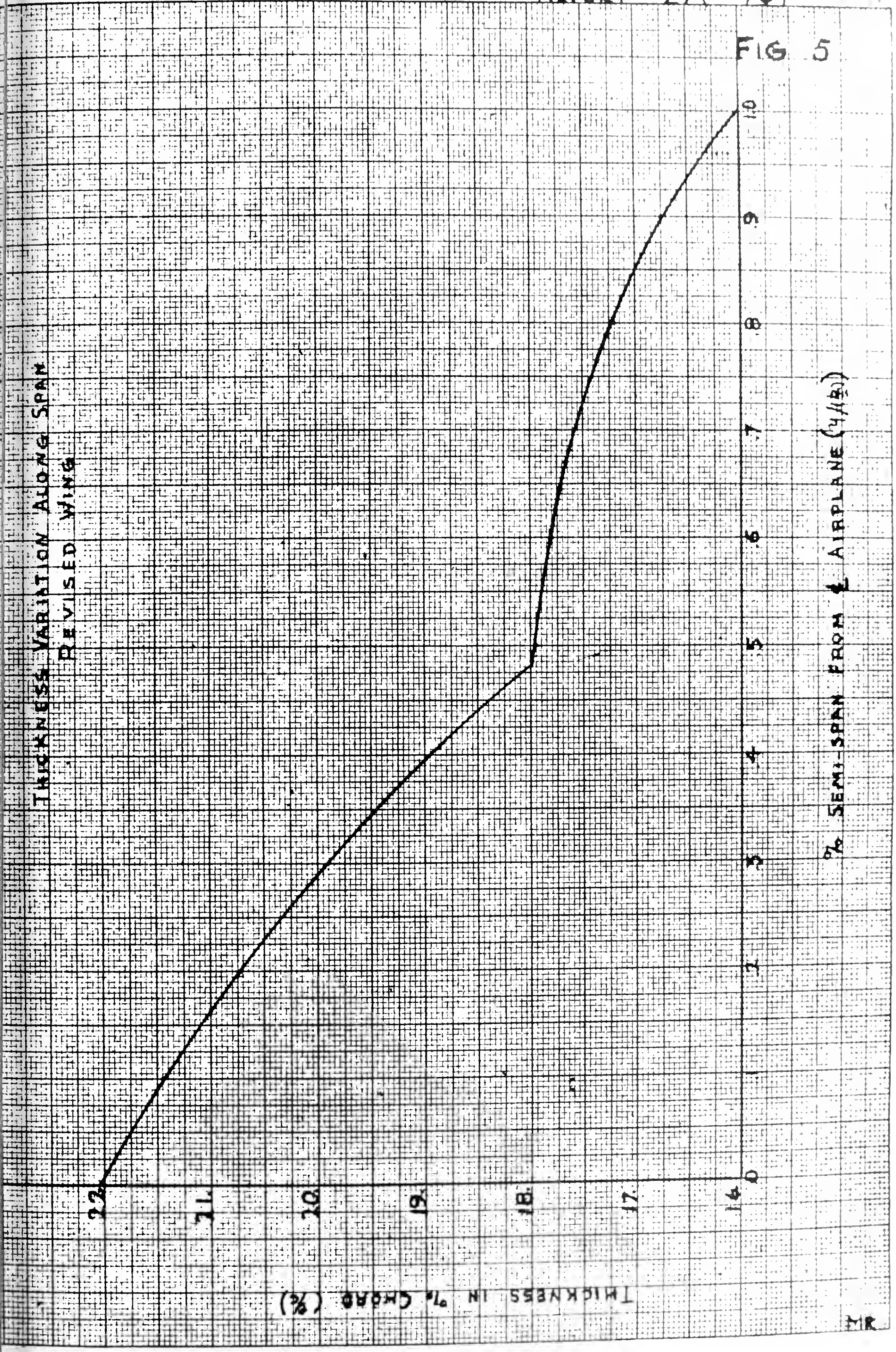


NOTE: $\Delta C_m = -.04$ ESTIMATED FOR FUSELAGE, NACELLES, ETC.



(236" R.C.)
FIG 4
CHORD VARIATION ALONG SPAN
REVISED WING
T.R. = 4.07
A.R. = 12.005
200
150
100
50
HORD
INCHES
(58" T.C.)
.2
.4
.6
.8
1.0
% SEMI-SPAN (b/2)
MR

FIG 5


THICKNESS VARIATION ALONG SPAN
REVISED WING
% SEMI-SPAN FROM ℄ AIRPLANE (y/(b/2))
THICKNESS IN % CHORD (%)
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
22
21
20
19
18
17
16
MR

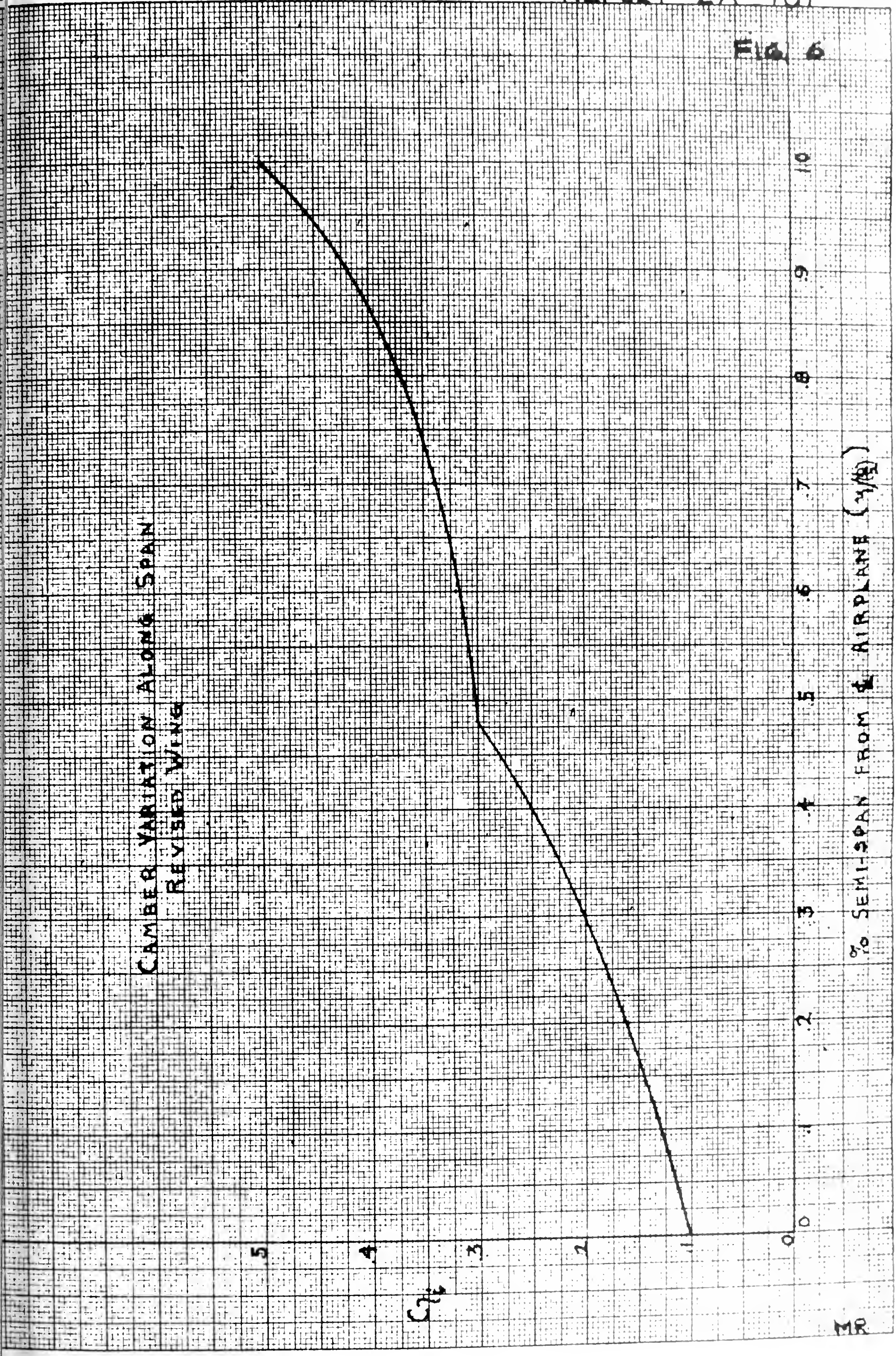
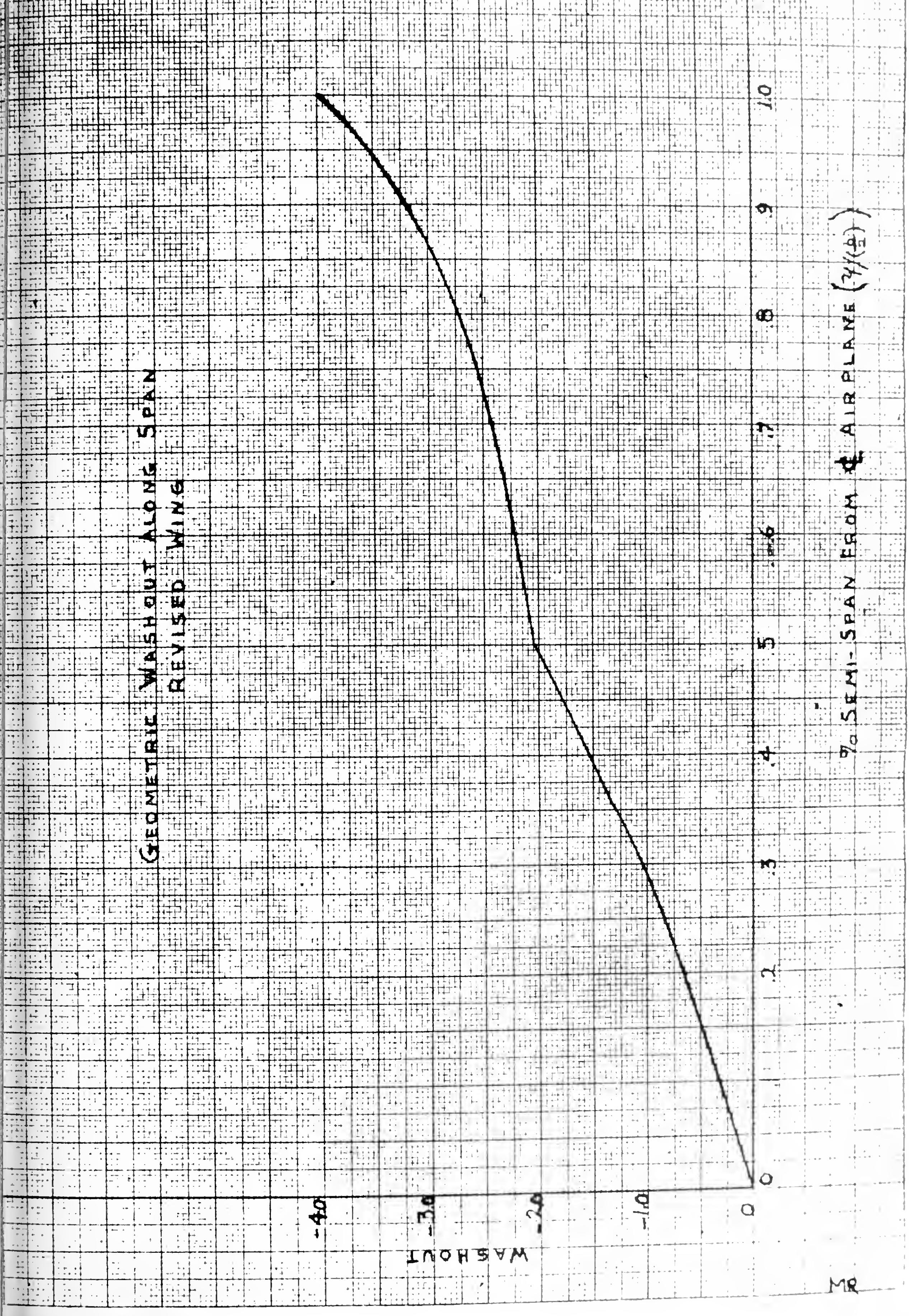

CAMBER VARIATION ALONG SPAN
REVISED WING
% SEMI-SPAN FROM ℄ AIRPLANE (y/b/2)
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
0
.1
.2
.3
.4
.5
C_{ℓ}
MR

FIG. 6

FIG 7


GEOMETRIC WASHOUT ALONG SPAN
REVISED WING
% SEMI-SPAN FROM ℄ AIRPLANE (y/(b/2))
WASHOUT
-4.0
-3.0
-2.0
-1.0
0
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
MR

APPENDIX A

OUTLINE OF PROCEDURE FOR THE CALCULATION OF AIRFOIL PRESSURE DISTRIBUTIONS

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OUTLINE OF PROCEDURE FOR THE

CALCULATION OF AIRFOIL PRESSURE DISTRIBUTIONS

s can be seen from the general theory there are three velocities etermine: $\frac{V_s}{V_0}$, the velocity due to the airfoil shape; $\frac{\Delta u}{V_0}$, the city increment due to camber (or to basic lift); and $\frac{\Delta V_a}{V_0}$, the city increment due to angle of attack (or to additional lift).

$$\frac{V_s}{V_0} = \left(\frac{V}{V_0} - 1\right)\frac{t}{t_b} + 1$$

re $\frac{V}{V_0}$ is the velocity on the base profile as given in Reference 2. s equation merely gives the correction to the velocity for thicknesses er than those of Reference 2.

To quote a numerical example (not related to the subject airfoils): t desired to determine the velocity on the surface of the symmetrical foil 63,4-022 at 0.25c

$$\left(\frac{V_s}{V_0}\right)_{0.25c} = (1.288 - 1)\frac{22}{20} + 1 = 1.317$$

This is also the velocity increment due to airfoil thickness at the face of a cambered airfoil at the same station.

$\frac{\Delta u}{V_0} C_{l_i}$ is the velocity increment due to the camber line. Since the be line can be replaced by a vortex sheet, the velocities will add on top of the airfoil and subtract on the bottom for positive lift.

$\frac{\Delta u}{V_0} C_{l_i} = \frac{P}{4}$ where P is the load at any point on the camber line. be approximated by the theoretical load, but should be obtained by egating the camber line slope:

$$\left(\frac{\Delta u}{V_0} C_{l_i}\right)_{\theta=\theta_0} = \left(\frac{P}{4}\right)_{\theta=\theta_0} = -\frac{1}{2\pi}\int_0^{2\pi} \left(\frac{dy_c}{d(x/c)}\right)\cot(\theta-\theta_0)\,d\theta$$

s ntegral can be evaluated numerically by the method given in 1 and 4 where F = $dy_c/d(x/c)$ in Reference 1).

procedure to be used is as follows:

Plot $dy_c/d\frac{x}{c}$ vs θ (not vs. $\frac{x}{c}$)

Fair the plot of $dy_c/d\frac{x}{c}$ vs. θ so that it intersects the line $\theta = 0$ (Theoretically $dy_c/d\frac{x}{c} = \infty$ at $\theta = 0$). This is illustrated in Fig. 3.

A rational method of estimating $\left(dy_c/d\frac{x}{c}\right)_{0,0}$ is shown in Appendix D.

Read off values of $dy_c/d\frac{x}{c}$ and $\frac{d}{d\theta}\left(dy_c/d\frac{x}{c}\right)$ at $\theta = 0$, 0.1π, 0.2π, ... 0.9π, π .

Set these values into the formula

$$\left(\frac{\Delta u}{V_0} C_{l_i}\right)_{\theta=\theta_0} = -\left\{a_0\left[\frac{d}{d\theta}\left(\frac{dy_c}{d\frac{x}{c}}\right)\right]_{\theta=\theta_0} + a_1\left[\left(dy_c/d\tfrac{x}{c}\right)_{\theta=\theta_0+0.1\pi} - \left(dy_c/d\tfrac{x}{c}\right)_{\theta=\theta_0-0.1\pi}\right] + \ldots\ldots + a_9\left[\left(dy_c/d\tfrac{x}{c}\right)_{\theta=\theta_0+0.9\pi} - \left(dy_c/d\tfrac{x}{c}\right)_{\theta=\theta_0-0.9\pi}\right]\right\}$$

numerical values of the coefficients a_n are:

$a_0 = 0.1000$	$a_5 = 0.0503$
$a_1 = 0.3473$	$a_6 = 0.0366$
$a_2 = 0.1572$	$a_7 = 0.0281$
$a_3 = 0.0996$	$a_8 = 0.0163$
$a_4 = 0.0691$	$a_9 = 0.0080$

$\Delta V_a/V_0\, C_{l_a}$ is the velocity increment due to the circulation around a symmetrical airfoil lifting at $C_l = C_{l_a}$ ($C_{l_a} = C_l - C_{l_b}$ where $C_{l_b} = 0$ for a symmetrical airfoil)

umerical values for $\left(\frac{\Delta V_a}{V_o}\right)_{C_{l_a}=1}$ are given in the same tables
hose for $\frac{V}{V_o}$. For any other value of C_{l_a} multiply the given
ue of $\left(\frac{\Delta V_a}{V_o}\right)_{C_{l_a}=1}$ by the desired value of C_{l_a} .

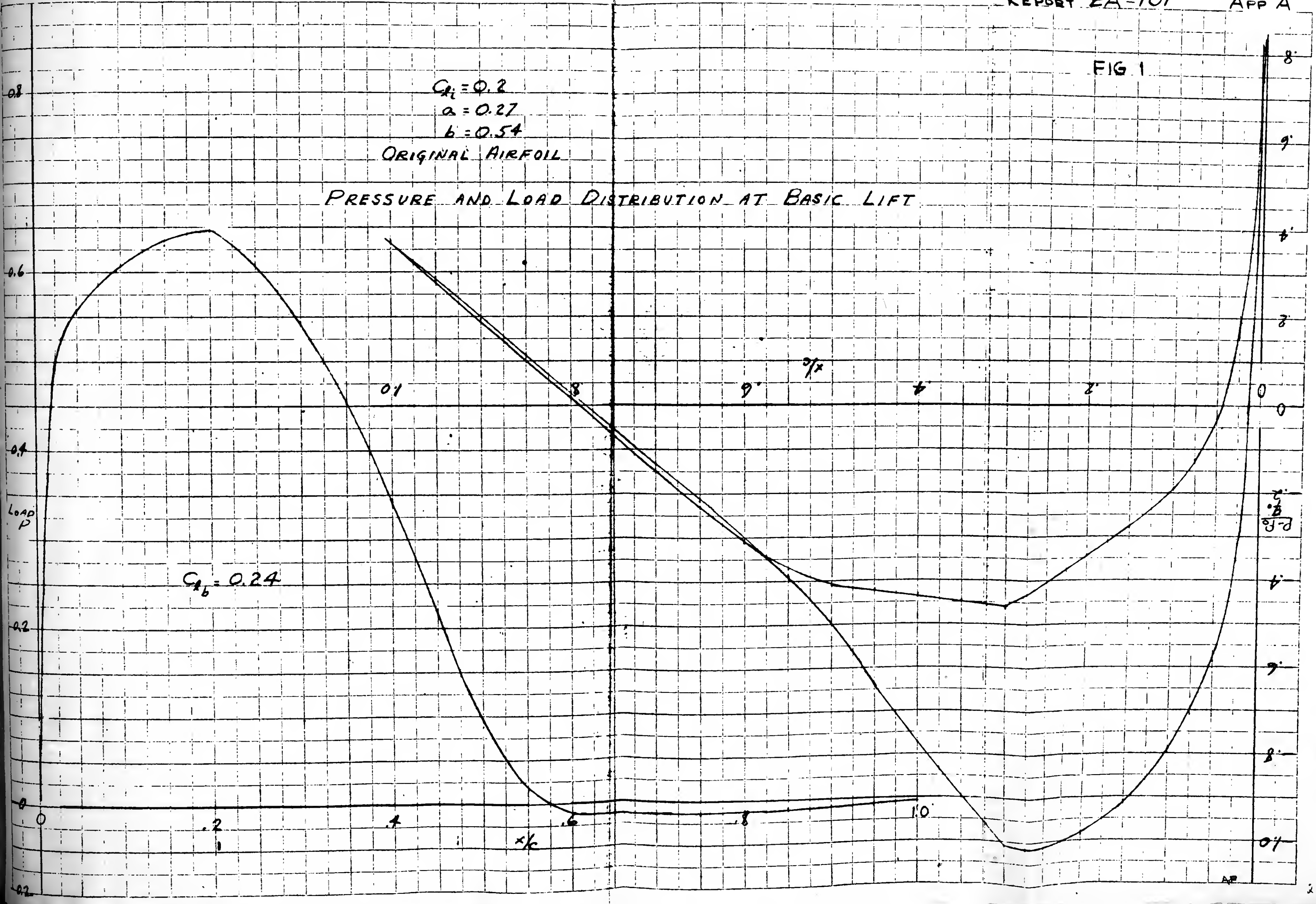


FIG 1
$C_{l_i} = 0.2$
$a = 0.27$
$b = 0.54$
ORIGINAL AIRFOIL
PRESSURE AND LOAD DISTRIBUTION AT BASIC LIFT
$C_{l_b} = 0.24$
LOAD P
x/c
0 .2 .4 .6 .8 1.0
0.8 0.6 0.4 0.2 0

FIG 2

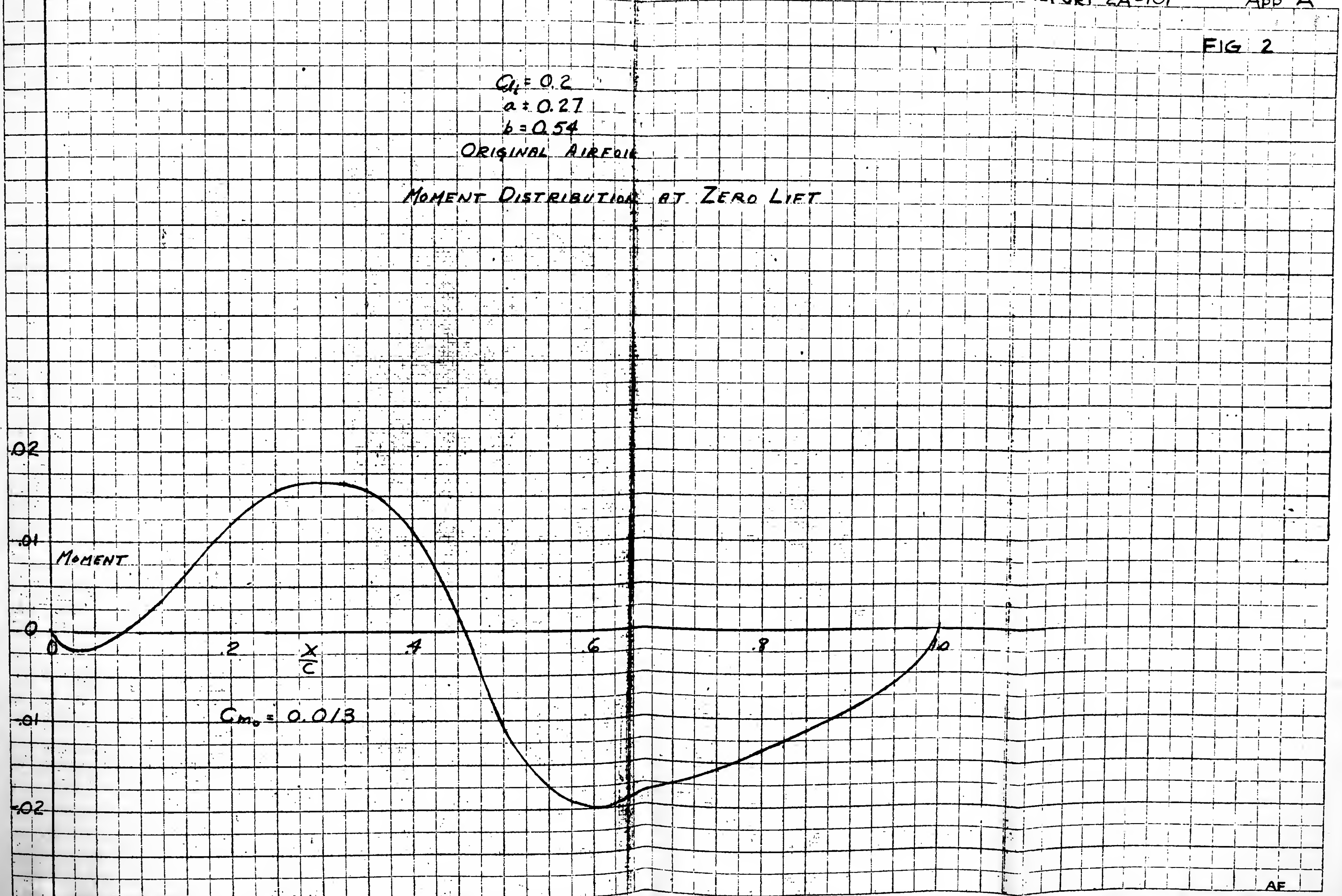
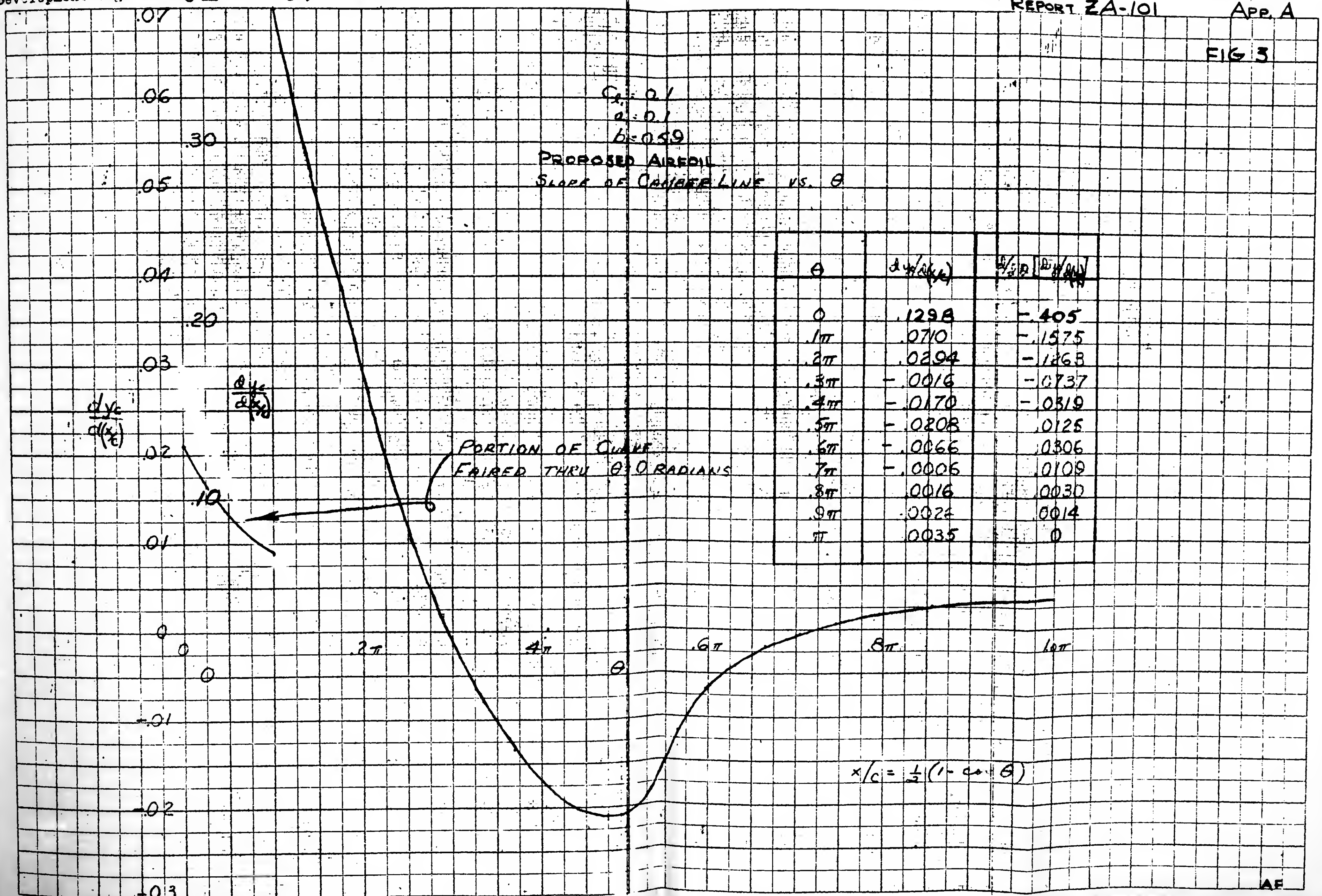

C_{l_i} = 0.2
a = 0.27
b = 0.54
ORIGINAL AIRFOIL
MOMENT DISTRIBUTION AT ZERO LIFT
.02
.01
MOMENT
0
-.01
-.02
0
.2
.4
.6
.8
1.0
$\frac{X}{C}$
C_{m_0} = 0.013
AF

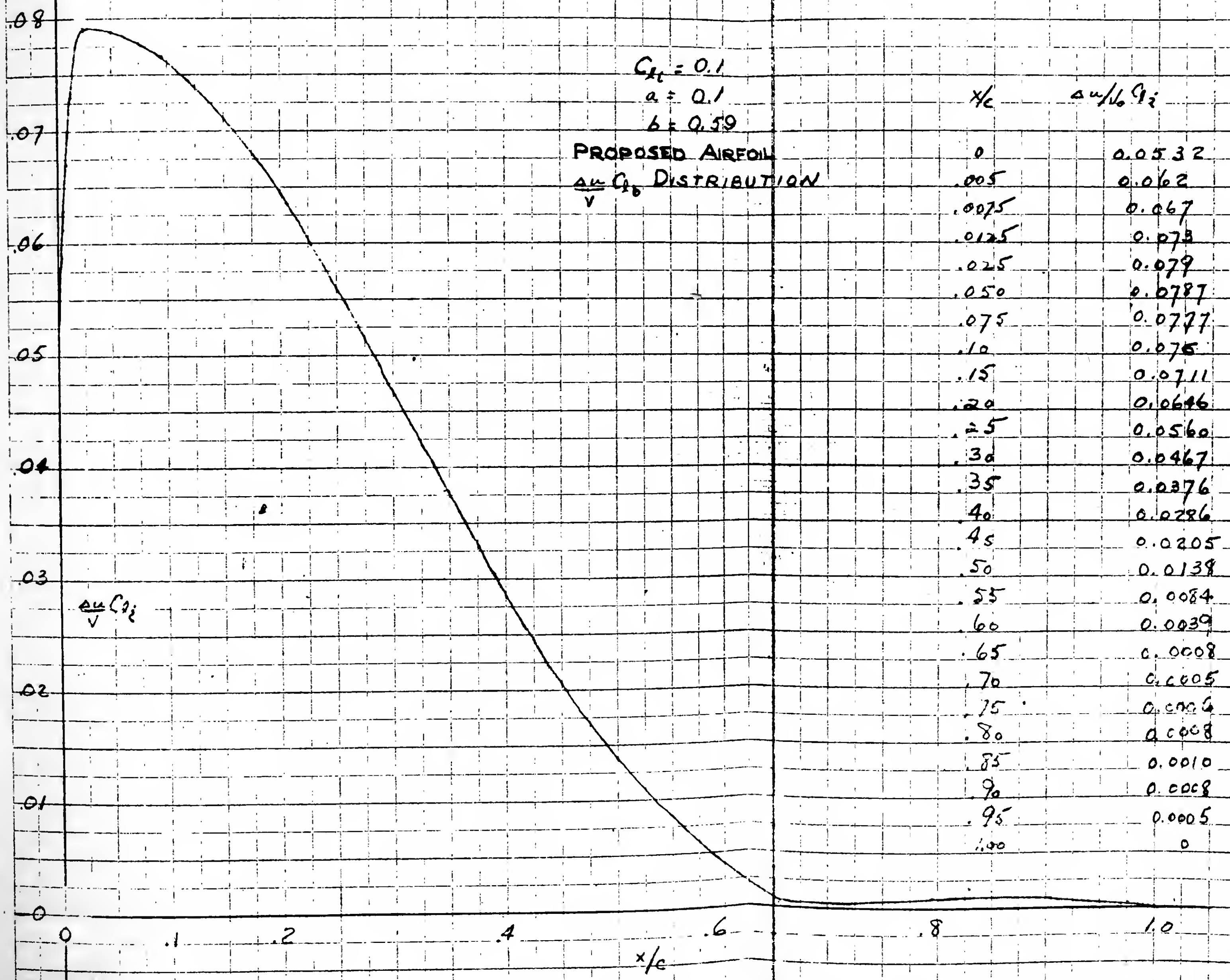
FIG 3



θ	$dy_c/d(x/c)$	$\frac{d}{d\theta}\left[\frac{dy_c}{d(x/c)}\right]$
0	.1298	-.405
.1π	.0710	-.1575
.2π	.0294	-.1268
.3π	-.0016	-.0737
.4π	-.0170	-.0319
.5π	-.0208	.0125
.6π	-.0066	.0306
.7π	-.0006	.0109
.8π	.0016	.0030
.9π	.0022	.0014
π	.0035	0

AF

FIG 4



x/c	$\Delta u/V_0 \, C_{l_i}$
0	0.0532
.005	0.062
.0075	0.067
.0125	0.073
.025	0.079
.050	0.0787
.075	0.0777
.10	0.075
.15	0.0711
.20	0.0646
.25	0.0560
.30	0.0467
.35	0.0376
.40	0.0286
.45	0.0205
.50	0.0138
.55	0.0084
.60	0.0039
.65	0.0008
.70	0.0005
.75	0.0006
.80	0.0008
.85	0.0010
.90	0.0008
.95	0.0005
1.00	0

AF

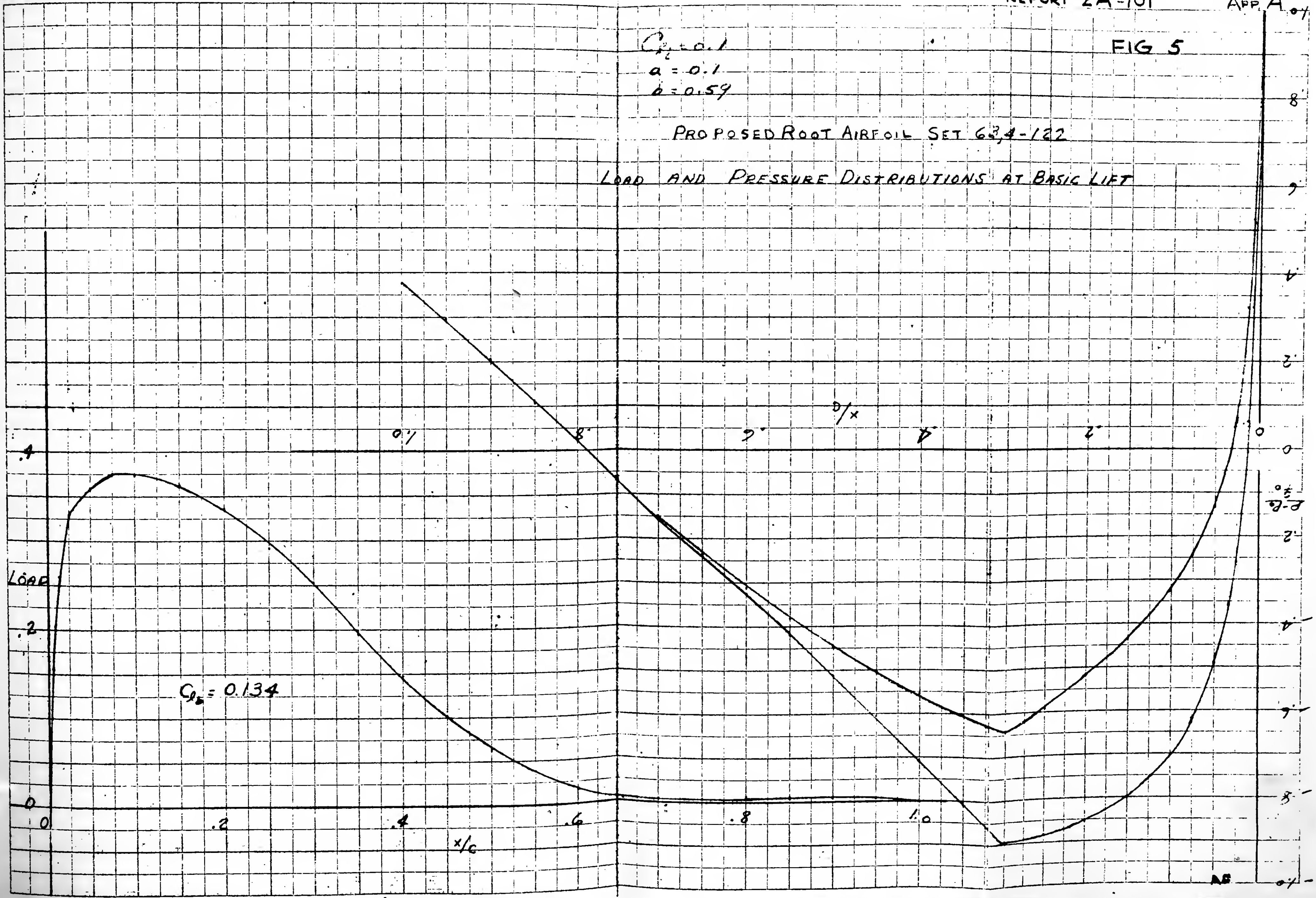
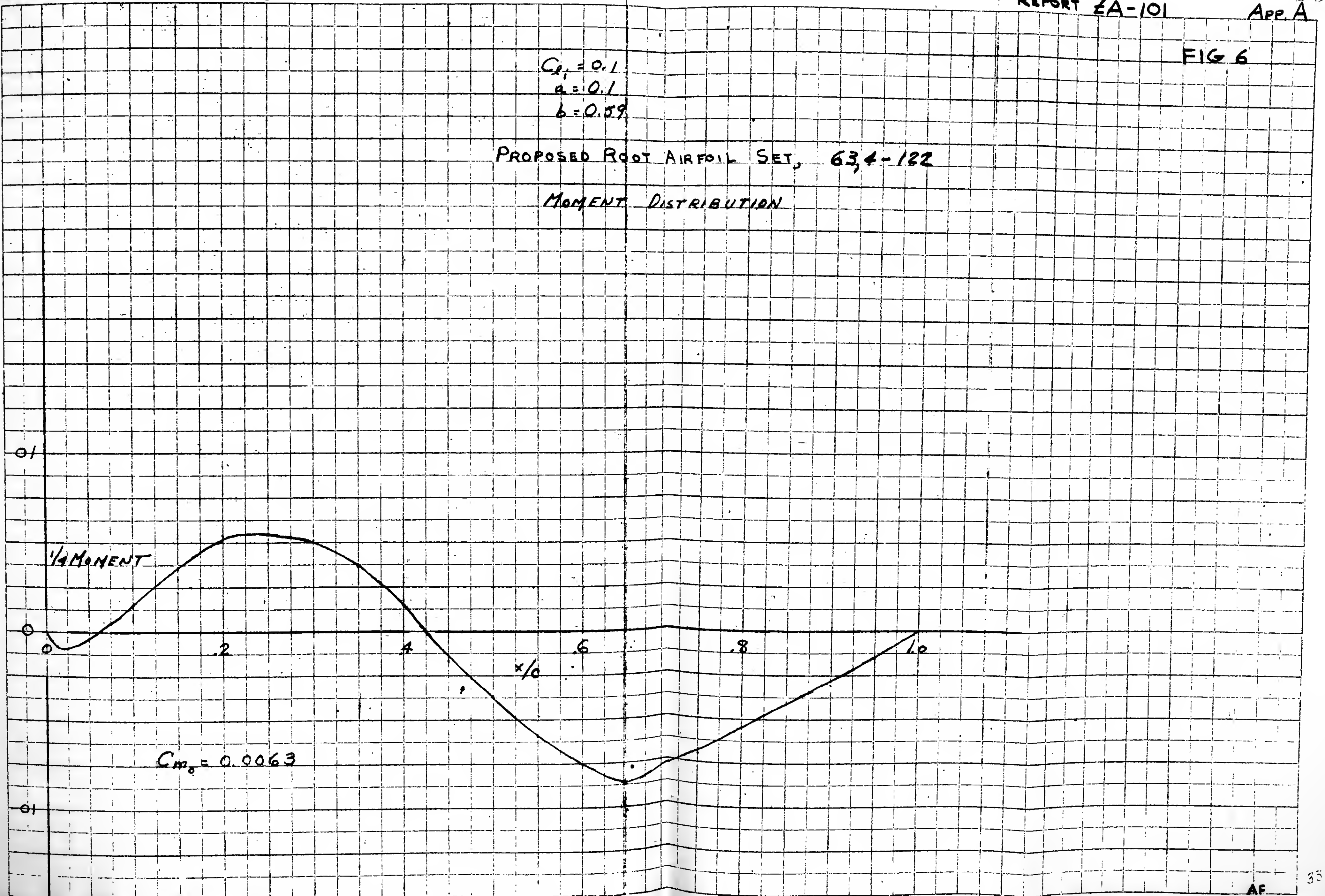

$C_{l_i} = 0.1$
$a = 0.1$
$b = 0.59$
FIG 5
PROPOSED ROOT AIRFOIL SET 63,4-122
LOAD AND PRESSURE DISTRIBUTIONS AT BASIC LIFT
LOAD
$C_{l_b} = 0.134$
x/c
0
.2
.4
.6
.8
1.0

FIG 6



$C_{l_i} = 0.1$
$a = 0.1$
$b = 0.59$
PROPOSED ROOT AIRFOIL SET, 63,4-122
MOMENT DISTRIBUTION
.01
1/4 MOMENT
0
0
.2
.4
.6
.8
1.0
x/c
$C_{m_o} = 0.0063$
-.01

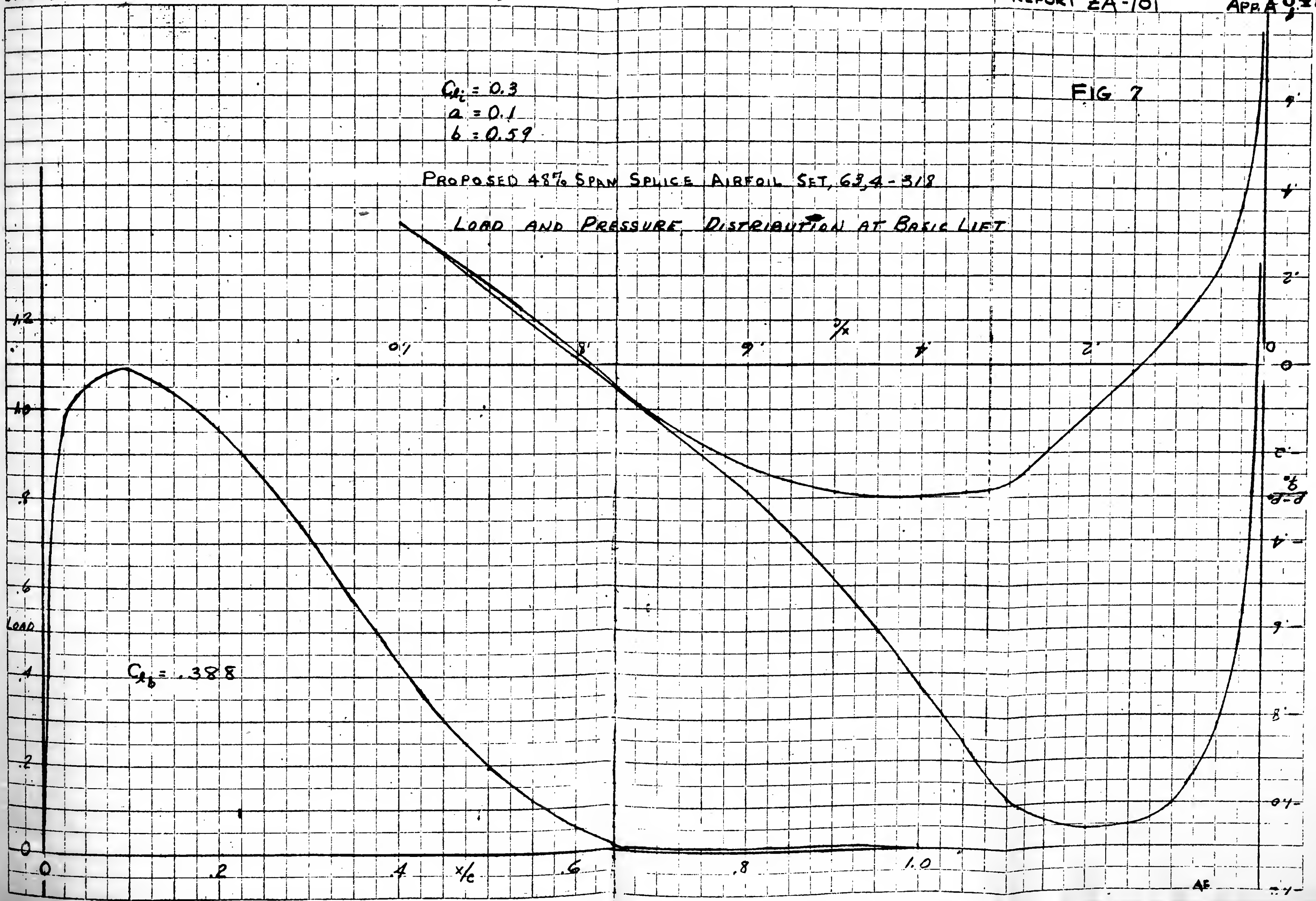


FIG 7
$C_{l_i} = 0.3$
$a = 0.1$
$b = 0.59$
PROPOSED 48% SPAN SPLICE AIRFOIL SET, 63,4-318
LOAD AND PRESSURE DISTRIBUTION AT BASIC LIFT
$C_{l_b} = .388$
LOAD
1.2
1.0
.8
.6
.4
.2
0
0
.2
.4
x/c
.6
.8
1.0

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Development Engineering . San Diego, California

FIG 8

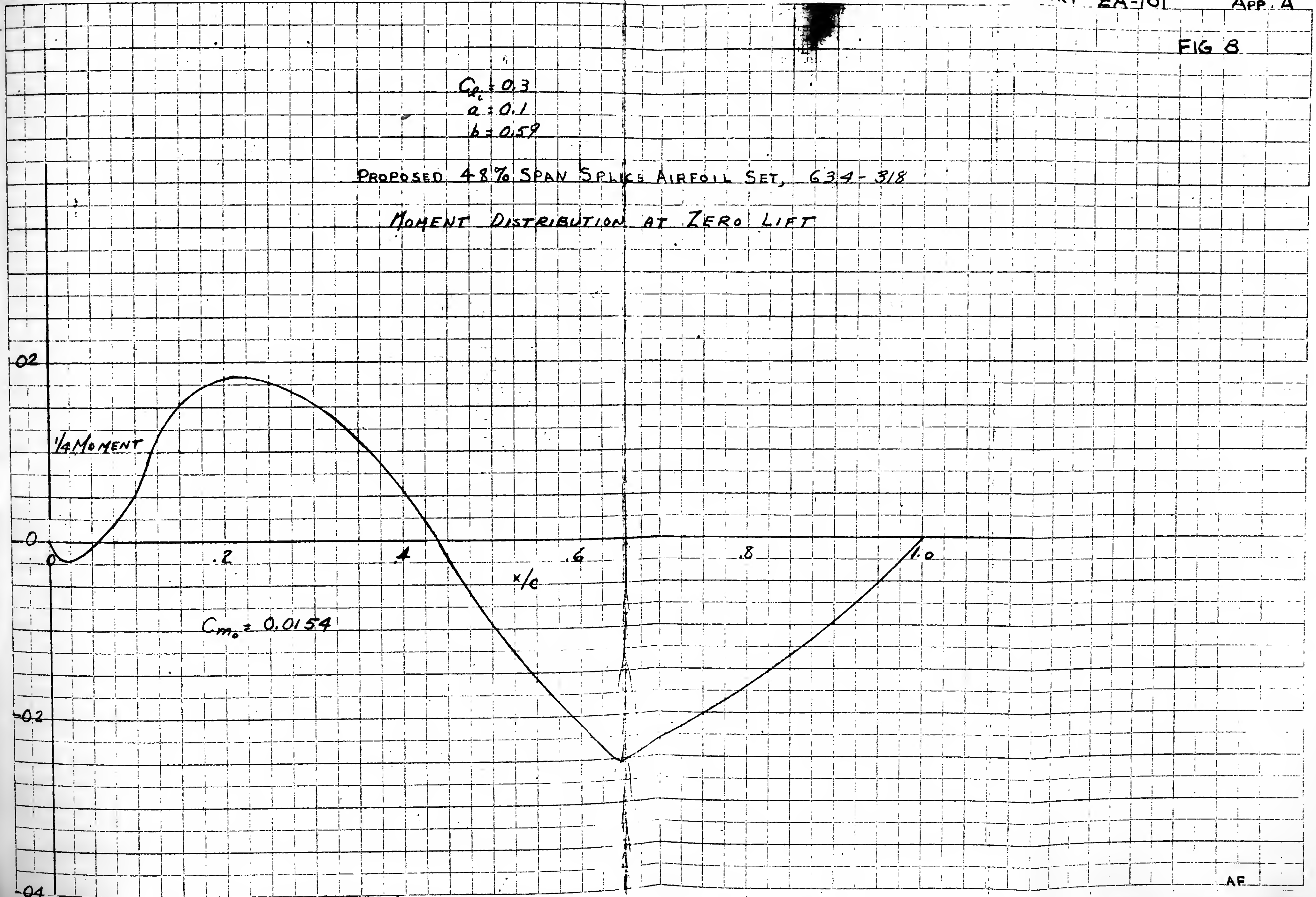


FIG. 9

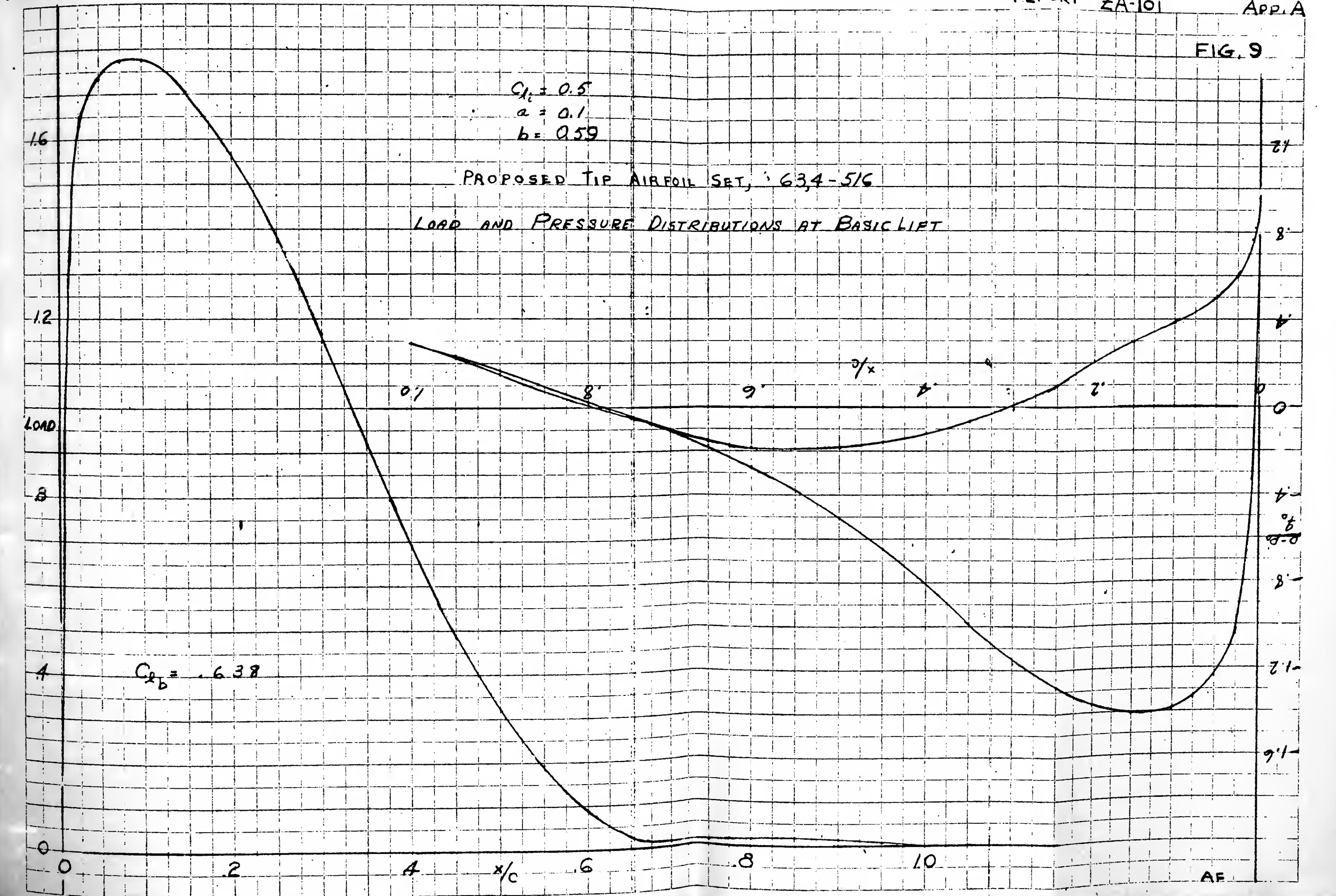
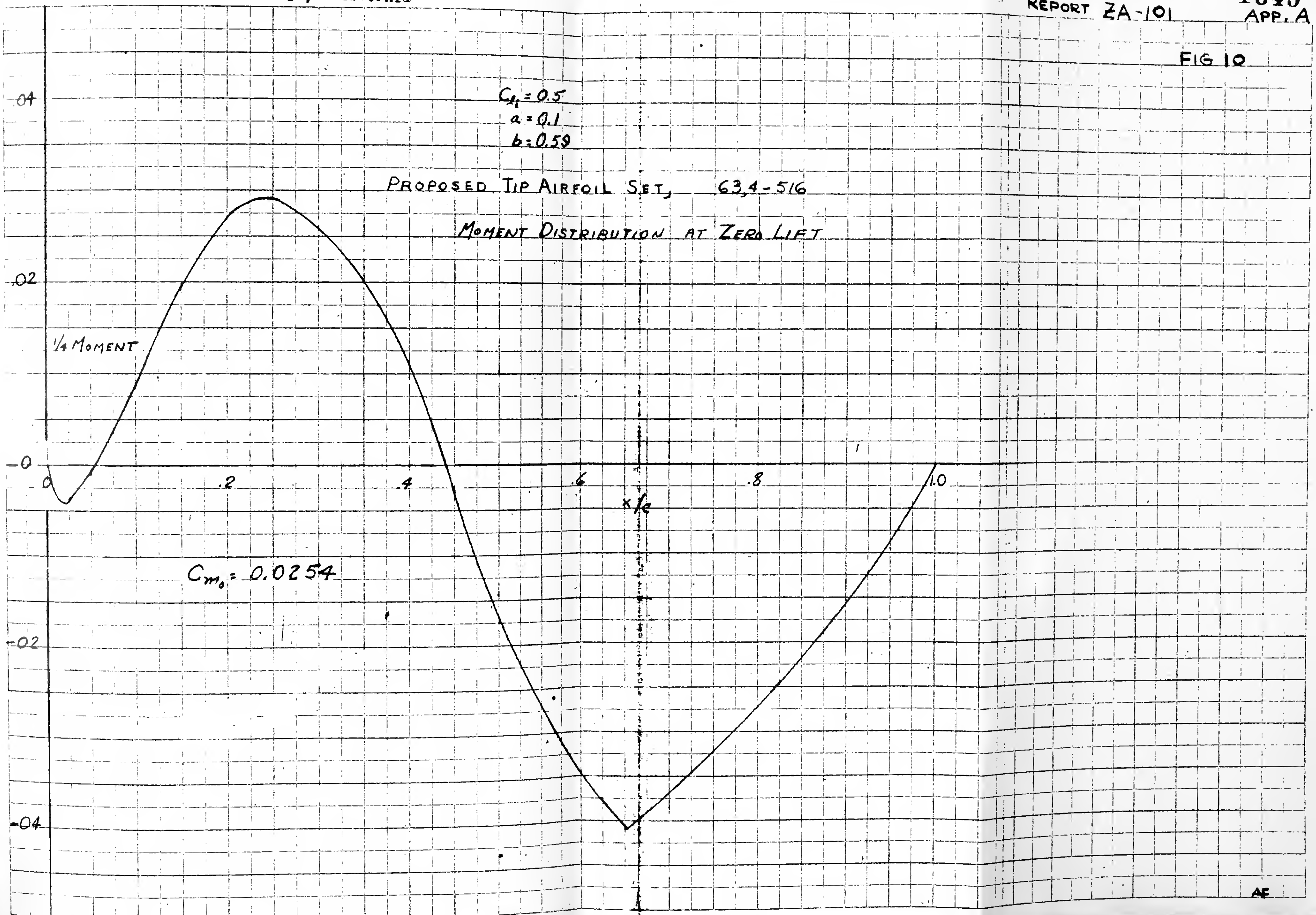

$C_{l_i} = 0.5$
$a = 0.1$
$b = 0.59$
PROPOSED TIP AIRFOIL SET, 63,4-516
LOAD AND PRESSURE DISTRIBUTIONS AT BASIC LIFT
1.6
1.2
LOAD
.8
.4
0
$C_{l_b} = .638$
0
.2
.4
x/c
.6
.8
1.0
AF

FIG 10

PROPOSED TIP AIRFOIL SET, 63,4-516

MOMENT DISTRIBUTION AT ZERO LIFT



INCREMENTAL VELOCITY DUE TO CAMBER

$C_{l_i} = 0.1 \quad (a = 0.1,\ b = 0.59)$

θ	$dy_c/d(x/c)$	$\frac{d(dy_c/dx)}{d\theta}$	1 a_n	θ = 0: 2	3	4 (3)-(2)	5 (1)(4)
0	.1665	-.532	0.1	-.532		.532	.0532
.1π	.0710	-.1575	.3473			0	0
.2π	.0294	-.1263	.1592				
.3π	-.0016	-.0737	.0996				
.4π	-.0170	-.0319	.0691				
.5π	-.0208	.0125	.0503				
.6π	-.0066	.0316	.0366				
.7π	-.0006	.0109	.0281				
.8π	.0016	.0030	.0163				
.9π	.0026	.0014	.0080				
π	.0034	0					

$\frac{\Delta u}{V_0} = .0532$

θ = .1π: 2	3	4 (3)-(2)	5 (1)(4)
-.1575		.1575	.0158
.0294	.1665	.1371	.0476
-.0016	.0710	.0726	.0114
-.0170	.0294	.0464	.0046
-.0208	-.0016	.0192	.0013
-.0066	-.0170	-.0104	-.0005
-.0006	-.0208	-.0202	-.0007
.0016	-.0066	-.0082	-.0002
.0026	-.0006	-.0032	-.00005
.0034	.0016	-.0018	-.00001

$\frac{\Delta u}{V_0} = +.0793$

θ = .2π: 2	3	4 (3)-(2)	5 (1)(4)
-.1263		.1263	.0126
-.0016	.0710	.0726	.0252
-.0170	.1665	.1835	.0288
-.0208	.0710	.0918	.0092
-.0066	.0294	.0360	.0025
-.0006	-.0016	-.0010	-.00005
.0016	-.0170	-.0186	-.00068
.0026	-.0208	-.0234	-.00066
.0034	-.0066	-.0100	-.00016
.0026	-.0006	-.0032	-.00002

$\frac{\Delta u}{V_0} = +.0767$

1	θ = .3π: 2	3	4	5
.1	-.0737		.0737	.0074
.3473	-.0170	.0294	.0464	.0162
.1572	-.0208	.0710	.0918	.0144
.0996	-.0066	.1665	.1731	.0173
.0691	-.0006	.0710	.0716	.0050
.0503	.0016	.0294	.0278	.0014
.0366	.0026	-.0016	-.0042	-.0002
.0281	.0034	-.0170	-.0204	-.0006
.0163	.0026	-.0208	-.0234	-.0004
.0080	.0016	-.0066	-.0082	-.0001

$\frac{\Delta u}{V_0} = +.0634$

θ = .4π: 2	3	4	5
-.0319		.0319	.0032
-.0208	-.0016	.0192	.0067
-.0066	.0294	.0360	.0057
-.0006	.0710	.0716	.0071
.0016	.1665	.1649	.0114
.0026	.0710	.0684	.0034
.0034	.0294	.0260	.0010
.0026	-.0016	-.0042	-.0001
.0016	-.0170	-.0186	-.0003
-.0006	-.0208	-.0202	-.0002

$\frac{\Delta u}{V_0} = +.0379$

θ = .5π: 2	3	4	5
.0125		-.0125	-.0013
-.0066	-.0170	-.0104	-.0036
-.0006	-.0016	-.0010	-.00016
.0016	.0294	+.0278	+.0028
.0026	.0710	.0684	.0047
.0034	.1665	.1631	.0082
.0026	.0710	.0684	.0035
.0016	.0294	.0278	.00078
-.0006	-.0016	-.0010	.00001
-.0066	-.0170	-.0164	.00008

$\frac{\Delta u}{V_0} = +.0138$

INCREMENTAL VELOCITY DUE TO CAMBER, $\frac{\Delta u}{V_0}$

$C_{l_i} = 0.1 \quad (a = 0.1,\ b = 0.59)$

TABLE II

$\theta = .6\pi$

1	2	3	4	5
.1	.0306		-.0306	-.0031
.3473	-.0006	-.0208	-.0202	-.0070
.1572	.0016	-.0170	-.0186	-.0029
.0996	.0026	-.0016	-.0042	-.0004
.0691	.0034	.0294	+.0260	+.0018
.0503	.0026	.0710	.0684	.0034
.0366	.0016	.1665	.1649	.0060
.0281	-.0006	.0710	.0716	.0020
.0163	-.0066	.0294	.0360	.00059
.0080	-.0208	-.0016	.0192	.00015

$\frac{\Delta u}{V_0} = +.0005$

$\theta = .7\pi$

2	3	4	5
.0109		-.0109	-.0011
.0016	-.0066	-.0082	-.0028
.0026	-.0208	-.0234	-.0037
.0034	-.0170	-.0204	-.0020
.0026	-.0016	-.0042	-.0003
.0016	.0294	+.0278	+.0014
-.0006	.0710	.0716	.0026
-.0066	.1665	.1731	.0049
-.0208	.0710	.0918	.0015
-.0170	.0294	.0464	.00037

$\frac{\Delta u}{V_0} = +.0009$

$\theta = .8\pi$

2	3	4	5
.0030		-.0030	-.0003
.0026	-.0006	-.0032	-.0011
.0034	-.0066	-.0100	-.0016
.0026	-.0208	-.0234	-.0023
.0016	-.0170	-.0186	-.0013
-.0006	-.0016	-.0010	-.00005
-.0066	.0294	+.0360	+.0013
-.0208	.0710	.0918	.0026
-.0170	.1665	.1835	.0030
-.0016	.0710	.0726	.00058

$\frac{\Delta u}{V_0} = +.0008$

$\theta = .9\pi$

1	2	3	4	5
.1	.0014		-.0014	-.0001
.3473	.0034	.0016	-.0018	-.0006
.1572	.0026	-.0006	-.0032	-.0005
.0996	.0016	-.0066	-.0082	-.0008
.0691	-.0006	-.0208	-.0202	-.0014
.0503	-.0066	-.0170	-.0104	-.0005
.0366	-.0208	-.0016	+.0192	+.0007
.0281	-.0170	.0294	.0464	.0019
.0163	-.0016	.0710	.0726	.0012
.0080	.0294	.1665	.1371	.0011

$\frac{\Delta u}{V_0} = +.0004$

$\theta = \pi$

1		4	5
0		0	0

$\frac{\Delta u}{V_0} = 0$

$$\left(\frac{\Delta u}{V_0} C_{l_i}\right)_{\theta=\theta_0} = -\left\{(a_0)\left[\frac{d}{d\theta}\left(\frac{dy_c}{d(\frac{x}{c})}\right)\right]_{\theta=\theta_0} + (a_1)\left[\left(\frac{dy_c}{d(\frac{x}{c})}\right)_{\theta=\theta_0+0.1\pi} - \left(\frac{dy_c}{d(\frac{x}{c})}\right)_{\theta=\theta_0-0.1\pi}\right]\right.$$
$$+ (a_2)\left[\left(\frac{dy_c}{d(\frac{x}{c})}\right)_{\theta=\theta_0+0.2\pi} - \left(\frac{dy_c}{d(\frac{x}{c})}\right)_{\theta=\theta_0-0.2\pi}\right] + \ldots\ldots$$
$$\left. + (a_9)\left[\left(\frac{dy_c}{d(\frac{x}{c})}\right)_{\theta=\theta_0+0.9\pi} - \left(\frac{dy_c}{d(\frac{x}{c})}\right)_{\theta=\theta_0-0.9\pi}\right]\right\}$$

CALCULATION OF PRESSURE DISTRIBUTION

PROPOSED ROOT SECTION
63,4-122 (a=0.1, b=0.59)

TABLE III

$C_{l_b} = .134$

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17
x/c	v/V_0	(2)-1	1.1(3)	v_t/V_0 (4)+1	$\Delta v/V_0\, C_{l}$	(5)+(6)	$(7)^2$	P_u 1-(8)	(5)-(6)	$(10)^2$	P_l 1-(11)	LOAD (12)-(9)	$(\Delta v_a/V_0)$ × (.134	(6)-(14)	1/4 LOAD (5)(15)	1/4 MOM (1)(16)
0	0	-1.000	-1.100	-.100	.053	-.047	.002	.998	-.[illegible]53	.023	.977	-.021	.187	-.134	.0134	0
.0050	.666	-.334	-.368	.632	.062	.694	.482	.518	.[illegible]70	.325	.675	.157	.172	-.110	-.0695	-.0004
.0075	.778	-.222	-.244	.756	.067	.823	.678	.322	.[illegible]89	.475	.525	.203	.161	-.094	-.0711	-.0005
.0125	.906	-.094	-.104	.896	.073	.969	.938	.062	.[illegible]23	.678	.322	.260	.144	-.071	-.0636	-.0008
.0250	1.039	.039	.043	1.043	.079	1.122	1.260	-.260	.[illegible]64	.929	.071	.331	.113	-.034	-.0355	-.0009
.0500	1.130	.130	.143	1.143	.079	1.222	1.492	-.492	1.[illegible]64	1.133	-.133	.359	.086	-.007	-.0080	-.0004
.0750	1.176	.176	.194	1.194	.078	1.272	1.620	-.620	1.[illegible]16	1.244	-.244	.376	.073	.005	.0060	.0005
.1000	1.207	.207	.228	1.228	.076	1.304	1.702	-.702	1.[illegible]52	1.328	-.328	.374	.064	.012	.0147	.0015
.1500	1.245	.245	.270	1.270	.071	1.341	1.800	-.800	1.1[illegible]9	1.436	-.436	.364	.052	.019	.0241	.0036
.2000	1.270	.270	.297	1.297	.065	1.362	1.856	-.856	1.2[illegible]	1.519	-.519	.337	.044	.021	.0272	.0054
.2500	1.288	.288	.317	1.317	.056	1.373	1.882	-.882	1.2[illegible]1	1.590	-.590	.292	.039	.017	.0224	.0055
.3000	1.300	.300	.330	1.330	.047	1.377	1.899	-.899	1.[illegible]83	1.645	-.645	.254	.024	.013	.0173	.0052
.3500	1.277	.277	.305	1.305	.037	1.342	1.802	-.802	1.[illegible]68	1.607	-.607	.195	.027	.008	.0105	.0037
.4000	1.252	.252	.277	1.277	.029	1.306	1.704	-.704	1.[illegible]48	1.556	-.556	.148	.026	.003	.0038	.0015
.4500	1.225	.225	.247	1.247	.020	1.267	1.607	-.607	1.2[illegible]7	1.505	-.505	.102	.023	-.003	-.0037	-.0017
.5000	1.197	.197	.217	1.217	.014	1.231	1.513	-.513	1.[illegible]03	1.450	-.450	.063	.020	-.006	-.0073	-.0037
.5500	1.167	.167	.184	1.184	.008	1.192	1.420	-.420	1.[illegible]6	1.381	-.381	.039	.017	-.009	-.0107	-.0059
.6000	1.135	.135	.149	1.149	.004	1.153	1.328	-.328	1.1[illegible]5	1.311	-.311	.017	.015	-.011	-.0126	-.0076
.6500	1.101	.101	.111	1.111	.001	1.112	1.236	-.236	1.1[illegible]0	1.232	-.232	.004	.013	-.012	-.0133	-.0087
.7000	1.066	.066	.073	1.073	.001	1.074	1.155	-.155	1.[illegible]72	1.150	-.150	.005	.011	-.010	-.0107	-.0075
.7500	1.029	.029	.032	1.032	.001	1.033	1.066	-.066	1.[illegible]31	1.062	-.062	.004	.010	-.009	-.0093	-.0070
.8000	.989	-.011	-.012	.988	.001	.989	.978	.022	.[illegible]87	.974	.974	.004	.008	-.007	-.0069	-.0055
.8500	.947	-.053	-.058	.942	.001	.943	.892	.108	.[illegible]1	.885	.885	.007	.006	-.005	-.0047	-.0040
.9000	.901	-.099	-.109	.891	.001	.892	.795	.205	.[illegible]0	.792	.792	.003	.005	-.004	-.0036	-.0032
.9500	.853	-.147	-.162	.838	.001	.839	.704	.296	.[illegible]7	.700	.700	.004	.003	-.002	-.0017	-.0016
1.0000	.807	-.193	-.212	.788	0	.788	.622	.378	.[illegible]8	.622	.622	0	0	0	0	0

$$v_t/V_0 = (v/V_0 - 1)\frac{t}{t_b} + 1$$

$$t/t_b = \frac{0.22}{0.20} = 1.10$$

TABLE IV

CALCULATION OF PRESSURE DISTRIBUTION
PROPOSED AIRFOIL @ 0.48 SEMI-SPAN
63,4-318 (a=0.1, b=0.59)

$C_{l_b} = 0.388$

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17
x/c	v/V_0	(2)-1	0.9(3)	V_t/V_0 (4)+1	$\Delta v/V_0 \, C_{l_i}$	(5)+(6)	$(7)^2$	P_u 1-(8)	(5)-(6)	$(10)^2$	P_l 1-(11)	LOAD (12)-(9)	$(\Delta V_a/V_0) \times (.388)$	(6)-(14)	1/4 LOAD (5)(15)	1/4 MOM. (1)(16)
0	0	-1.000	-.900	.100	.160	.260	.067	-.933	-.060	.004	.996	.063	.542	-.382	-.038	0
.0050	.666	-.334	-.301	.699	.186	.885	.783	.217	.513	.264	.736	.519	.497	-.311	-.218	-.0011
.0075	.778	-.222	-.200	.800	.201	1.001	1.002	-.002	.599	.359	.641	.643	.467	-.266	-.213	-.0016
.0125	.906	-.094	-.085	.915	.219	1.134	1.289	-.289	.696	.485	.515	.804	.416	-.197	-.180	-.0020
.0250	1.039	.039	.035	1.035	.237	1.272	1.620	-.620	.798	.638	.362	.982	.328	-.091	-.094	-.0023
.0500	1.130	.130	.117	1.117	.236	1.353	1.830	-.830	.881	.777	.223	1.053	.250	-.014	-.016	-.0008
.0750	1.176	.176	.158	1.158	.233	1.391	1.935	-.935	.925	.855	.145	1.080	.211	.022	.025	.0019
.1000	1.207	.207	.186	1.186	.228	1.414	2.000	-1.000	.958	.919	.091	1.091	.184	.044	.052	.0052
.1500	1.245	.245	.220	1.220	.213	1.433	2.050	-1.050	1.007	1.014	-.014	1.036	.150	.063	.077	.0155
.2000	1.270	.270	.243	1.243	.194	1.437	2.062	-1.062	1.049	1.100	-.100	.962	.128	.066	.082	.0164
.2500	1.288	.288	.259	1.259	.168	1.427	2.040	-1.040	1.091	1.190	-.190	.850	.112	.056	.071	.0177
.3000	1.300	.300	.270	1.270	.140	1.410	1.990	-.990	1.130	1.278	-.278	.712	.100	.040	.051	.0153
.3500	1.277	.277	.249	1.249	.113	1.362	1.855	-.855	1.136	1.288	-.288	.567	.085	.028	.035	.0123
.4000	1.252	.252	.227	1.227	.086	1.313	1.727	-.727	1.141	1.303	-.303	.424	.074	.012	.015	.0060
.4500	1.225	.225	.202	1.202	.062	1.264	1.601	-.601	1.140	1.300	-.300	.301	.066	-.004	-.005	-.0023
.5000	1.197	.197	.177	1.177	.041	1.218	1.482	-.482	1.136	1.290	-.290	.192	.057	-.016	-.019	-.0095
.5500	1.167	.167	.150	1.150	.025	1.175	1.381	-.381	1.125	1.265	-.265	.116	.050	-.025	-.029	-.0160
.6000	1.135	.135	.121	1.121	.012	1.133	1.282	-.282	1.109	1.230	-.230	.052	.043	-.031	-.035	-.0210
.6500	1.101	.101	.091	1.091	.002	1.093	1.192	-.192	1.089	1.185	-.185	.007	.038	-.036	-.039	-.0254
.7000	1.066	.066	.059	1.059	.002	1.061	1.126	-.126	1.057	1.119	-.119	.007	.033	-.031	-.033	-.0231
.7500	1.029	.029	.026	1.026	.002	1.028	1.057	-.057	1.024	1.050	-.050	.007	.028	-.026	-.027	-.0202
.8000	.989	-.011	-.010	.990	.002	.992	.983	.017	.988	.976	.024	.007	.023	-.021	-.021	-.0168
.8500	.947	-.053	-.048	.952	.003	.955	.913	.087	.949	.901	.099	.012	.018	-.015	-.014	-.0119
.9000	.901	-.099	-.089	.911	.002	.913	.835	.165	.909	.826	.174	.009	.014	-.012	-.011	-.0099
.9500	.853	-.147	-.132	.868	.002	.870	.757	.243	.866	.750	.250	.007	.009	-.007	-.006	-.0057
1.0000	.807	-.193	-.174	.826	0	.826	.682	.318	.826	.682	.318	0	0	0	0	0

$$V_t/V_0 = (v/V_0 - 1)\frac{t}{t_b} + 1$$

$$t/t_b = \frac{0.18}{0.20} = 0.90$$

CALCULATION OF PRESSURE DISTRIBUTION
PROPOSED TIP SECTION
63,4-516 ($a=0.1, b=0.59$)

TABLE V

$C_{\ell_b} = 0.638$

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17
x/c	V/V_0	V/V_0-1	0.8(3)	V_t/V 1+(4)	$\frac{\Delta u}{V_0} C_{\ell_i}$	(5)+(6)	$(7)^2$	P_u 1-(8)	(5)-(6)	$(10)^2$	P_ℓ 1-(11)	LOAD (12)-(9)	$(\Delta v_a/V_0)$ × (.638)	(6)-(14)	1/4 LOAD (3)(15)	1/4 MOM. (1)(16)
0	0	-1.	-.800	.200	.266	.466	.217	.783	-.066	.004	.996	.213	.891	-.625	-.125	0
.0050	.666	-.334	-.267	.733	.310	1.043	1.090	-.090	.423	.180	.820	.910	.817	-.507	-.372	-.0019
.0075	.778	-.222	-.178	.822	.335	1.157	1.338	-.338	.487	.237	.763	1.101	.768	-.433	-.356	-.0027
.0125	.906	-.094	-.075	.925	.365	1.290	1.665	-.665	.560	.314	.686	1.351	.685	-.320	-.296	-.0037
.0250	1.039	.039	.031	1.031	.395	1.426	2.030	-1.030	.636	.404	.596	1.626	.540	-.145	-.150	-.0038
.0500	1.130	.130	.104	1.104	.394	1.498	2.240	-1.240	.710	.504	.496	1.736	.412	-.018	-.020	-.0010
.0750	1.176	.176	.141	1.141	.389	1.530	2.340	-1.340	.752	.565	.435	1.775	.347	.042	.048	.0036
.1000	1.207	.207	.166	1.166	.380	1.546	2.390	-1.390	.786	.618	.382	1.772	.303	.077	.090	.0090
.1500	1.245	.245	.196	1.196	.356	1.552	2.410	-1.410	.840	.706	.294	1.704	.246	.110	.132	.0198
.2000	1.270	.270	.216	1.216	.323	1.539	2.362	-1.362	.893	.798	.202	1.564	.211	.112	.136	.0272
.2500	1.288	.288	.231	1.231	.280	1.511	2.280	-1.280	.951	.904	.096	1.376	.185	.095	.117	.0292
.3000	1.300	.300	.240	1.240	.234	1.474	2.175	-1.175	1.006	1.012	-.012	1.163	.164	.070	.089	.0261
.3500	1.277	.277	.222	1.222	.188	1.410	1.989	-.989	1.034	1.070	-.070	.919	.140	.048	.059	.0207
.4000	1.252	.252	.202	1.202	.143	1.345	1.812	-.812	1.059	1.120	-.120	.692	.123	.020	.024	.0096
.4500	1.225	.225	.180	1.180	.103	1.283	1.645	-.645	1.077	1.160	-.160	.485	.108	-.005	-.006	-.0027
.5000	1.197	.197	.158	1.158	.069	1.227	1.507	-.507	1.089	1.185	-.185	.322	.095	-.026	-.030	-.0150
.5500	1.167	.167	.134	1.134	.042	1.176	1.382	-.382	1.092	1.191	-.191	.191	.082	-.040	-.045	-.0247
.6000	1.135	.135	.108	1.108	.020	1.128	1.271	-.271	1.088	1.183	-.183	.088	.072	-.052	-.058	-.0348
.6500	1.101	.101	.081	1.081	.004	1.085	1.177	-.177	1.077	1.160	-.160	.017	.062	-.058	-.063	-.0409
.7000	1.066	.066	.053	1.053	.002	1.055	1.115	-.115	1.051	1.105	-.105	.010	.054	-.052	-.055	-.0385
.7500	1.029	.029	.023	1.023	.003	1.026	1.050	-.050	1.020	1.040	-.040	.010	.045	-.042	-.043	-.0322
.8000	.989	-.011	-.009	.991	.004	.995	.990	.010	.987	.973	.027	.017	.038	-.034	-.034	-.0272
.8500	.947	-.053	-.042	.958	.005	.963	.928	.072	.953	.909	.091	.017	.029	-.024	-.023	-.0196
.9000	.901	-.099	-.079	.921	.004	.925	.856	.144	.917	.840	.160	.016	.023	-.019	-.017	-.0153
.9500	.853	-.147	-.118	.882	.002	.884	.782	.218	.880	.775	.225	.007	.014	-.012	-.010	-.0095
1.0000	.807	-.193	-.155	.845	0	.845	.714	.286	.845	.714	.286	0	0	0	0	0

$$\frac{V_t}{V_0} = \left(\frac{V}{V_0}-1\right)\frac{t}{t_b}+1$$

$$\frac{t}{t_b} = \frac{0.16}{0.20} = 0.80$$

Defendants' Exhibit A—(Continued)

Consolidated Vultee Aircraft Corporation
San Diego Division

Model......Airplane Report No. ZA-101, App. B

Appendix B

Proposed Airfoil Ordinates & Profiles

Consolidated Vultee Aircraft Corporation
San Diego Division

Model......Airplane Report No. ZA-101, App, B

Airfoil Ordinates

Airfoil and camber line ordinates were calculated by the method outlined in Reference 7. These calculations are summarized in Table I. Airfoil profiles calculated by the above procedure are given in Figures 1 and 2.

FIG 1

ROOT SECTION
63,4, 122
a=.1 b=.59

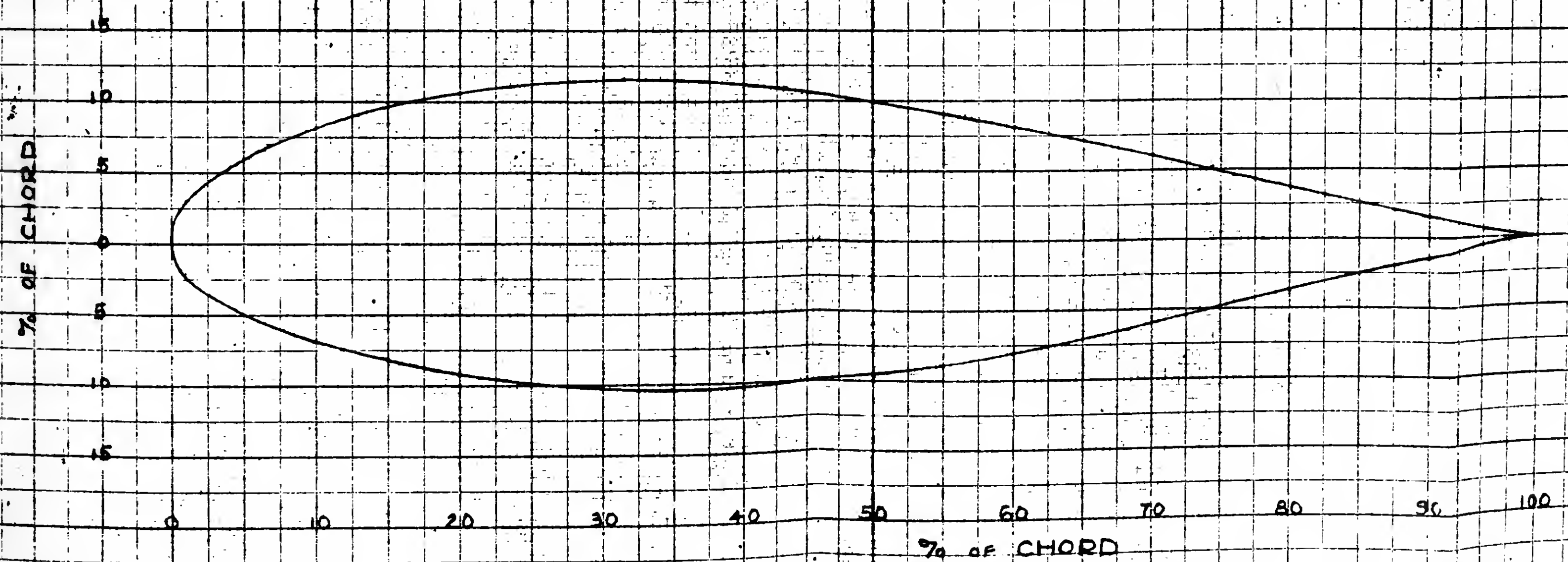
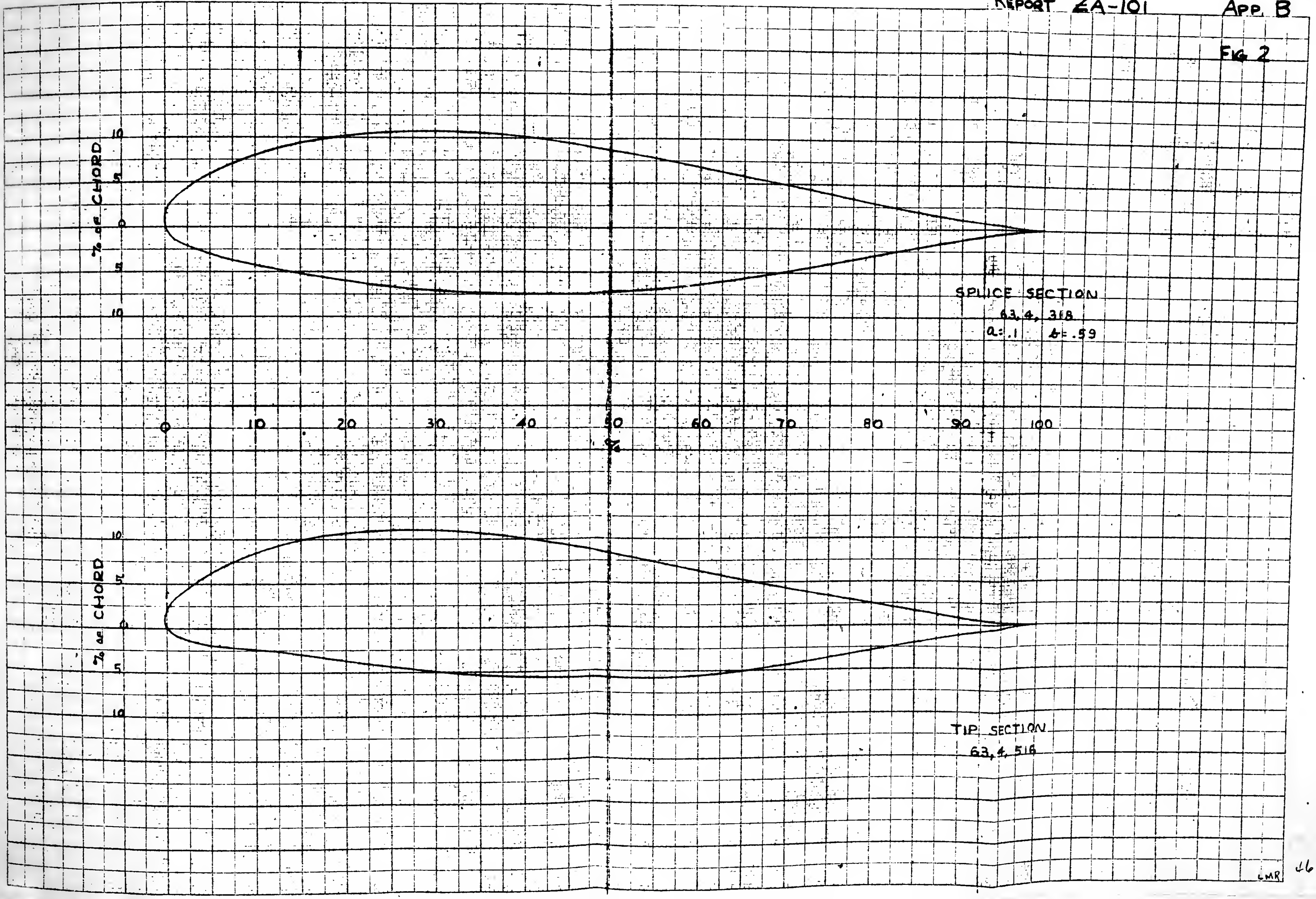


FIG 2


% of CHORD
10
5
0
5
10
SPLICE SECTION
63,4, 318
a=.1 b=.59
0
10
20
30
40
50
60
70
80
90
100
%
% of CHORD
10
5
0
5
10
TIP SECTION
63,4, 516
LMR

ORDINATES FOR PROPOSED AIRFOILS
REVISED WING

TABLE I

ROOT — 63_4-122 (a=0.1, b=0.59)

X	Yc	dyc/dx	Xu	Yu	Xℓ	Yℓ
0	0		0	0	0	0
.005	.000579	.108701	.00296	.01953	.00704	-.01616
.0075	.000919	.099130	.00524	.02370	.00976	-.02186
.0125	.001382	.086913	.00929	.03029	.01501	-.02753
.025	.002350	.069777	.02224	.04186	.02776	-.03716
.050	.003840	.051216	.04722	.05819	.05278	-.05051
.075	.004958	.038777	.07246	.07051	.07754	-.06059
.10	.005788	.027386	.09785	.08056	.10205	-.06898
.15	.006711	.010432	.14907	.09570	.15093	-.08228
.20	.006925	-.000273	.20003	.10600	.19997	-.09214
.25	.006759	-.007957	.25084	.11269	.24916	-.09917
.30	.006164	-.013459	.30147	.11566	.29853	-.10334
.35	.005381	-.017493	.35192	.11512	.34808	-.10436
.40	.004439	-.019992	.40215	.11184	.39785	-.10296
.45	.003405	-.021106	.45217	.10642	.44783	-.09960
.50	.002353	-.020737	.50201	.09934	.49799	-.09464
.55	.001363	-.018457	.55165	.09091	.54835	-.08819
.60	.000578	-.011635	.60094	.08143	.59906	-.08027
.65	.000130	-.006866	.65049	.07123	.64951	-.07097
.70	-.000138	-.004042	.70024	.06032	.69976	-.06060
.75	-.000287	-.001921	.75009	.04884	.74991	-.04942
.80	-.000347	-.000454	.80002	.03706	.79999	-.03776
.85	-.000337	.000808	.84998	.02542	.85002	-.02610
.90	-.000270	.001847	.89997	.01546	.90003	-.01510
.95	-.000155	.002723	.94998	.00535	.95002	-.00567
1.00	0	.003473	1.00000	0	1.00000	0

0.48 SEMI-SPAN — 63_4-318 (a=0.1, b=0.59)

X	Yc	dyc/dx	Xu	Yu	Xℓ	Yℓ
0	0		0	0	0	0
.005	.001737	.326103	.00022	.01640	.00978	-.01293
.0075	.002757	.297390	.00216	.02071	.01284	-.01519
.0125	.004146	.2[illegible]789	.00651	.02712	.01849	-.01883
.025	.007050	.21[illegible]	.01805	.03875	.03195	-.02475
.050	.011520	.153[illegible]	.04324	.05553	.05676	-.03249
.075	.014874	.116[illegible]51	.06885	.06819	.08115	-.03644
.10	.017364	[illegible]	.09499	.07816	.10501	-.04343
.15	.020133	.031296	.14772	.09291	.15228	-.05264
.20	.020775	-.00[illegible]19	.20007	.10183	.19993	-.06028
.25	.019905	-.02[illegible]871	.25207	.10653	.24793	-.06674
.30	.018492	-.040577	.30362	.10801	.29639	-.07103
.35	.016143	-.052479	.35471	.10582	.34529	-.07354
.40	.013314	-.05[illegible]76	.40526	.10108	.39474	-.07425
.45	.010215	-.063518	.45533	.09434	.44467	-.07391
.50	.007059	-.062211	.50492	.08628	.49508	-.07216
.55	.004089	-.05[illegible]71	.55332	.07730	.54668	-.06912
.60	.001734	-.034905	.60231	.06785	.59769	-.06439
.65	.000390	-.020598	.65120	.05855	.64880	-.05777
.70	-.000414	-.012128	.70060	.04905	.69940	-.04987
.75	-.000861	-.00[illegible]765	.75023	.03933	.74977	-.04105
.80	-.001041	-.001562	.80004	.02957	.79996	-.03165
.85	-.001011	.002424	.84995	.02007	.85005	-.02209
.90	-.000810	.003541	.89993	.01132	.90007	-.01294
.95	-.000465	.006169	.94996	.00404	.95004	-.00497
1.00	0	.01[illegible]19	1.00000	0	1.00000	0

TIP — 63_4-516 (a=0.1, b=0.59)

X	Yc	dyc/dx	Xu	Yu	Xℓ	Yℓ
0	0		0	0	0	0
.005	.002895	.543505	-.00155	.01495	.01155	-.00915
.0075	.004595	.495650	.00011	.01952	.01489	-.01032
.0125	.006910	.434565	.00409	.02626	.02091	-.01244
.025	.011750	.348885	.01550	.03899	.03430	-.01549
.050	.019200	.256080	.04018	.05754	.05982	-.01914
.075	.024790	.193885	.06592	.07163	.08408	-.02205
.10	.028940	.136930	.09262	.08264	.10738	-.02496
.15	.033555	.052100	.14658	.09819	.15342	-.03107
.20	.034625	-.001365	.20010	.10668	.19990	-.03742
.25	.033795	-.039785	.25306	.11078	.24694	-.04318
.30	.030820	-.067255	.30535	.11028	.29465	-.04864
.35	.026905	-.087465	.35965	.10643	.34305	-.05261
.40	.022195	-.099960	.40777	.09993	.39223	-.05554
.45	.017025	-.105530	.45786	.09155	.44214	-.05749
.50	.011765	-.103685	.50728	.08194	.49272	-.05840
.55	.006815	-.092285	.55599	.071675	.54401	-.05604
.60	.002890	-.058175	.60342	.06160	.59658	-.05582
.65	.000650	-.034330	.65177	.05233	.64823	-.05103
.70	-.000690	-.020210	.70089	.04237	.69911	-.04465
.75	-.001435	-.009605	.75034	.03429	.74966	-.03717
.80	-.001735	-.002270	.80006	.02547	.79994	-.02895
.85	-.001685	.004040	.84992	.01705	.85008	-.02043
.90	-.001350	.009235	.89990	.00943	.90010	-.01213
.95	-.000775	.013616	.94995	.00323	.95005	-.00479
1.00	0	.017365	1.00000	0	1.00000	0

y_c – CAMBER LINE ORDINATES

dy_c/dx – SLOPE OF CAMBER LINE

X_u, Y_u – COORDINATES OF UPPER SURFACE POINT

X_ℓ, Y_ℓ – COORDINATES OF LOWER SURFACE POINT

Defendants' Exhibit A—(Continued)

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Consolidated Vultee Aircraft Corporation
San Diego Division

Model......Airplane Report No. ZA-101, App. C

Appendix C

Span-Load Distributions

SPAN-LOAD CALCULATIONS

Span Load calculations for elevator zero and elevator 10° are given in Tables I to IV. A graphical estimation of the stalling characterisitcs of the revised wing for these elevator conditions is given in Figures 1 and 2.

Pitching-moment calculations for the revised wing, elevator zero, were made by the method employed in ZA-056. For the estimation of full-scale results a correlation factor of 3 was applied to the calculated value given in Table II. As these calculations were made for a bare wing a ΔC_m of -.04 was added to these values for the estimation of complete model characteristics. Previous wind-tunnel tests of a tailless design show that this pitching-moment increment is a fair average for the change in pitching-moment due to the addition of fuselage, nacelles, etc.

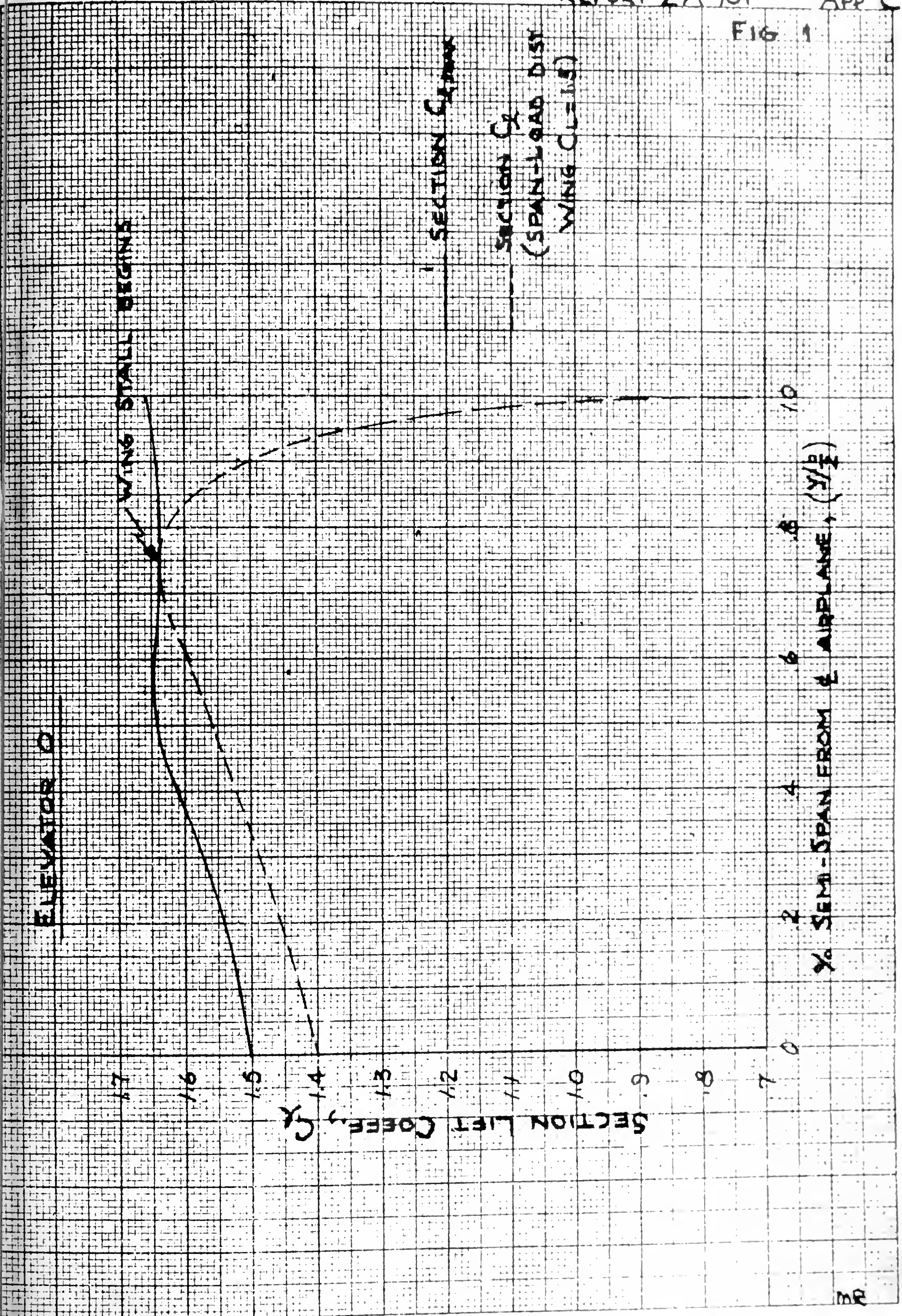
BY

CHECKED

APPROVED

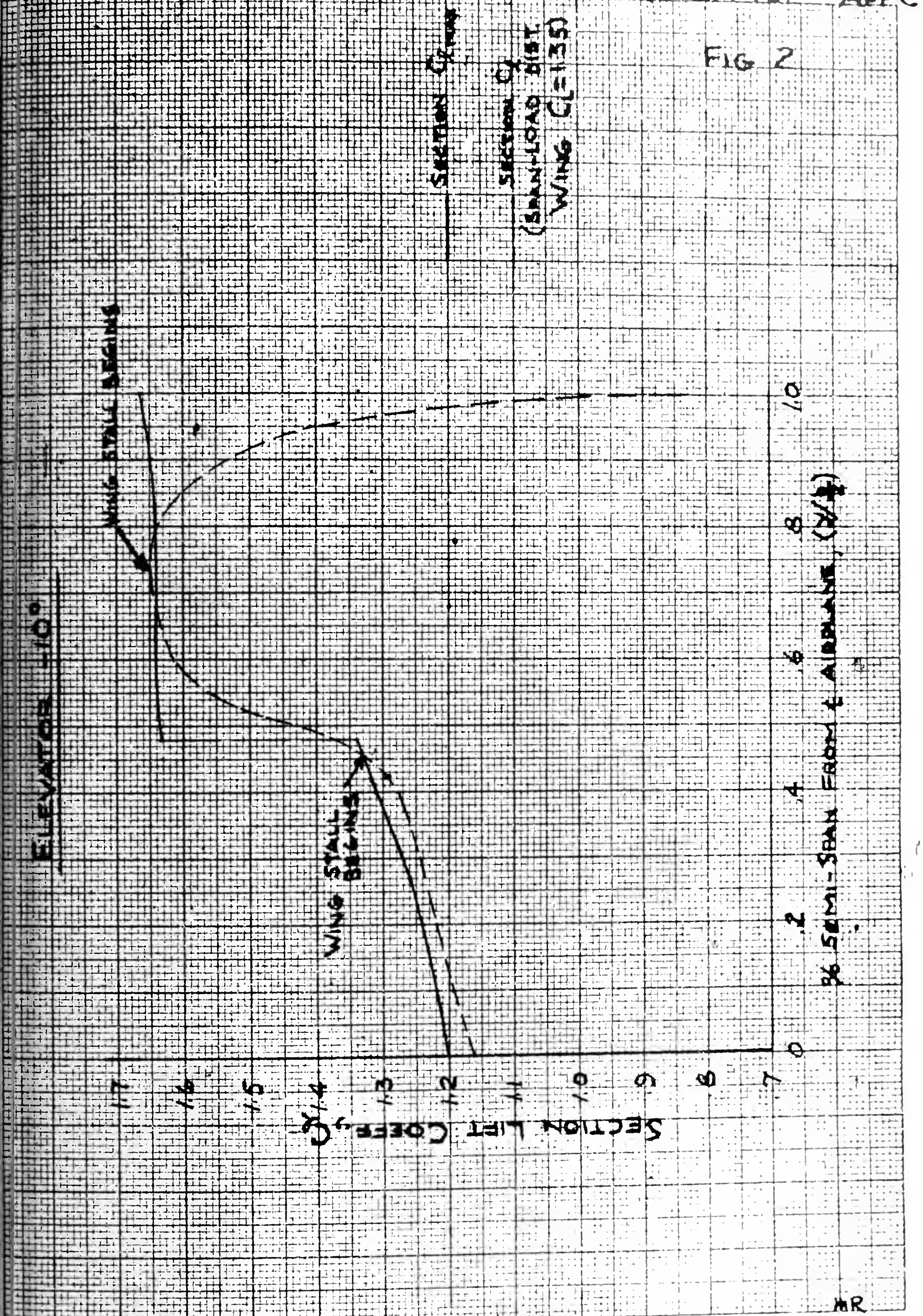
1058

FIG 1


WING STALL BEGINS
SECTION $C_{l_{max}}$
SECTION C_l
(SPAN-LOAD DIST
WING $C_L = 1.5$)
ELEVATOR 0
% SEMI-SPAN FROM ℄ AIRPLANE, $(y/\frac{b}{2})$
0
.2
.4
.6
.8
1.0
SECTION LIFT COEFF., C_l
1.7
1.6
1.5
1.4
1.3
1.2
1.1
1.0
.9
.8
.7

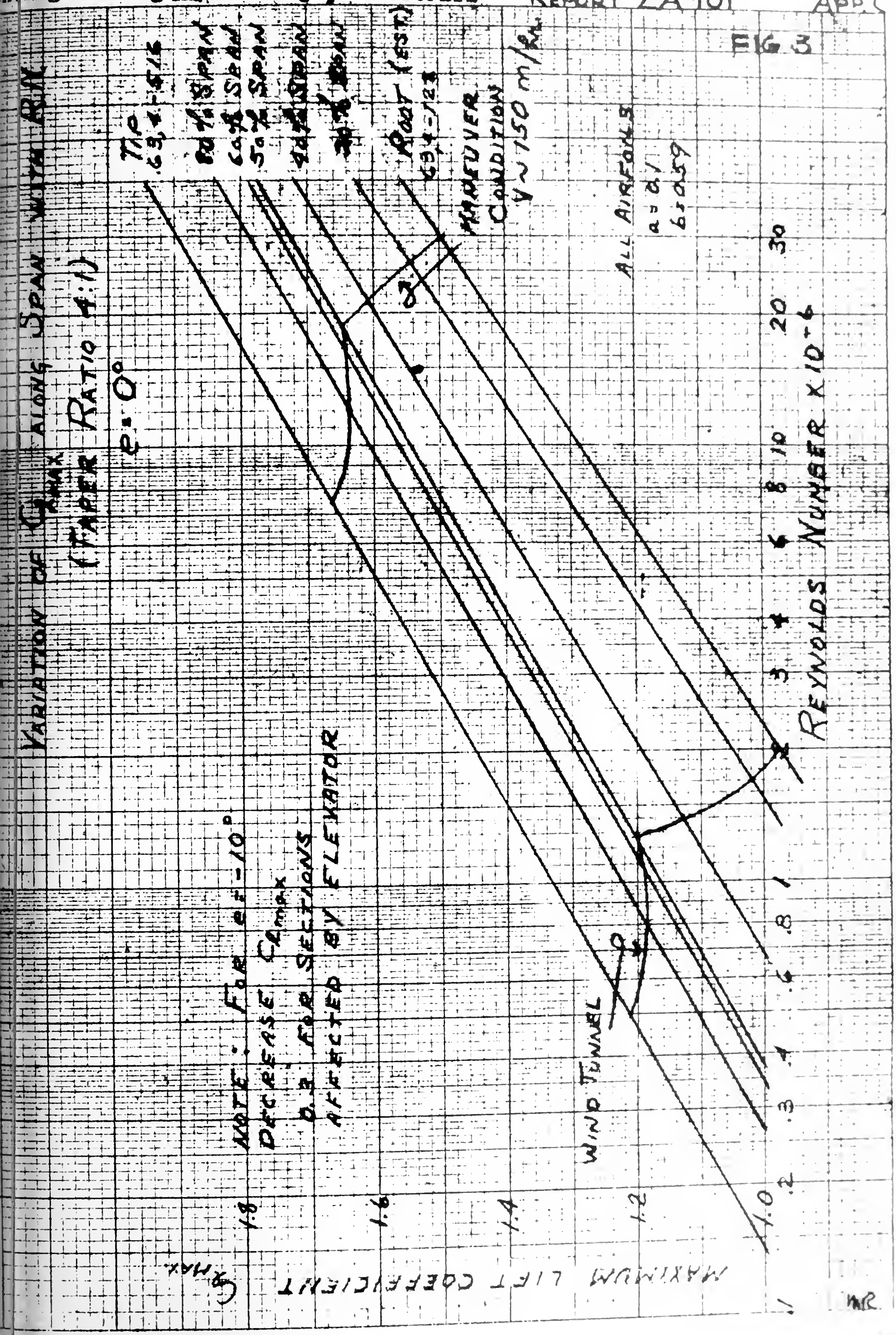
MR

FIG 2


ELEVATOR -10°
WING STALL BEGINS
WING STALL BEGINS
SECTION $C_{l_{max}}$
SECTION C_l
(SPAN-LOAD DIST)
WING $C_L = 1.35$
% SEMI-SPAN FROM ℄ AIRPLANE, (y/b/2)
0
.2
.4
.6
.8
1.0
SECTION LIFT COEFF, C_l
1.7
1.6
1.5
1.4
1.3
1.2
1.1
1.0
.9
.8
.7

MR

FIG 3


VARIATION OF $C_{L\,MAX}$ ALONG SPAN WITH R.N.
(TAPER RATIO 4:1)
ε = 0°
TIP
ROOT (EST)
MANEUVER CONDITION
V ~ 150 m/hr.
ALL AIRFOILS
NOTE: FOR $e_F = -10°$ DECREASE $C_{L max}$ 0.2 FOR SECTIONS AFFECTED BY ELEVATOR
WIND TUNNEL
REYNOLDS NUMBER × 10^{-6}
MAXIMUM LIFT COEFFICIENT
$C_{L\,MAX}$
1.8
1.6
1.4
1.2
1.0
.1
.2
.3
.4
.6
.8
1
2
3
4
6
8
10
20
30
WR

SPAN-LOAD DISTRIBUTION : 2-ENG. TAILLESS DESIGN

REPORT ZA-101 APP. C

REVISED AIRFOILS & WING

TABLE I

ELEV. 0°

$$C_{la_1} = \frac{1}{2}\left[\frac{a_0}{\bar{a}_0} + \frac{4\bar{c}}{\pi c}\sqrt{1-\left(\frac{y}{b/2}\right)^2}\right] \qquad C_{l_b} = \frac{a_0}{2}\left(\alpha_{R_0} + \beta\right)$$

1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17	18	19	20
$y/\frac{b}{2}$	CHORD C (in)	ΔY in.	MULT. ×3	$\int c\,dy$ (2)×(3)×(4)	a_0 ($dC_l/d\alpha$)	$a_0/\bar{a}_0$	$4\bar{c}/\pi c$	$[y/\frac{b}{2}]^2$	$1-[y/\frac{b}{2}]^2$	$\sqrt{1-[y/\frac{b}{2}]^2}$ $\sqrt{(10)}$	$4\bar{c}/\pi c \times \sqrt{1-[y/\frac{b}{2}]^2}$ (8)×(11)	(7)+(12)	C_{la_1} (13)/2	ε WASHOUT (DEG.)	α_{l_0}	β $(\varepsilon-\alpha_{l_0})$ (15)−(16)	$\int\beta c\,dy$ (17)×(5)	$\beta+\alpha_{R_0}$	C_{l_b} $(a_0/2)\times(19)$
0	236.0	88.20	1	20800	.120	1.0	.794	0	1.000	1.000	.794	1.794	.897	0	−.200	.200	4160	.813	.049
.100	218.0	↓	4	77000	↓	↓	.860	.010	.990	.995	.855	1.855	.927	−.30	−.270	−.030	−2310	.583	.025
.200	200.0		2	35300			.938	.040	.960	.980	.918	1.918	.959	−.64	−.350	−.290	−10250	.323	.019
.300	183.0		4	64500			1.022	.090	.910	.954	.980	1.980	.990	−1.04	−.500	−.540	−34800	.073	.004
.400	165.0		2	29100			1.134	.160	.840	.915	1.042	2.042	1.021	−1.54	−.730	−.810	−23600	−.197	−.012
.500	147.0	↓	4	51800			1.274	.250	.750	.865	1.102	2.102	1.051	−2.04	−1.060	−.980	−50700	−.367	−.022
.600	129.0	88.20	1	11370			1.450	.360	.640	.800	1.160	2.160	1.080	−2.20	−1.170	−1.030	−11700	−.417	−.025
.600	129.0	44.10	1	5685			1.450	.360	.640	.800	1.160	2.160	1.080	−2.20	−1.170	−1.030	−5850	−.417	−.025
.650	120.0	↓	4	21200			1.560	.420	.580	.760	1.186	2.186	1.093	−2.30	−1.250	−1.050	−22300	−.437	−.026
.700	111.0		2	9780			1.690	.490	.510	.714	1.210	2.210	1.105	−2.42	−1.340	−1.080	−10550	−.467	−.028
.750	102.0		4	18000			1.840	.560	.440	.663	1.220	2.220	1.110	−2.56	−1.450	−1.110	−19980	−.497	−.030
.800	94.0		2	8290			1.990	.640	.360	.600	1.190	2.190	1.095	−2.74	−1.600	−1.140	−9450	−.527	−.032
.850	85.0	↓	4	15000			2.200	.720	.280	.529	1.162	2.162	1.081	−2.95	−1.780	−1.170	−17550	−.557	−.034
.900	76.0	44.10	1	3350			2.460	.810	.190	.436	1.070	2.070	1.035	−3.20	−2.060	−1.140	−3820	−.527	−.032
.900	76.0	22.05	1	1675			2.460	.810	.190	.436	1.070	2.070	1.035	−3.20	−2.060	−1.140	−1910	−.527	−.032
.925	71.5	↓	4	6310			2.620	.855	.145	.381	1.000	2.000	1.000	−3.35	−2.225	−1.125	−7100	−.512	−.031
.950	67.0		2	2960			2.800	.905	.095	.308	.863	1.863	.931	−3.55	−2.430	−1.120	−3320	−.507	−.031
.975	62.5	↓	4	5510	↓	↓	3.000	.950	.050	.224	.672	1.672	.836	−3.76	−2.650	−1.111	−6110	−.497	−.030
1.000	58.0	22.05	1	1280			3.230	1.000	0	0	0	1.000	.500	−4.00	−2.980	−1.020	−1305	−.407	−.025

$\Sigma = 388910$

$$\begin{array}{r} 4160 \\ -242605 \\ \hline \Sigma\ -238445 \end{array}$$

$S = \frac{2}{3} \times 388910/144 = 1800\ \square'$

$\alpha_{R_0} = -\frac{1}{(\frac{b}{2})\bar{c}}\int_0^{\frac{b}{2}} \beta c\,dy = -\left(\frac{-238445}{388{,}910}\right) = +.613°$

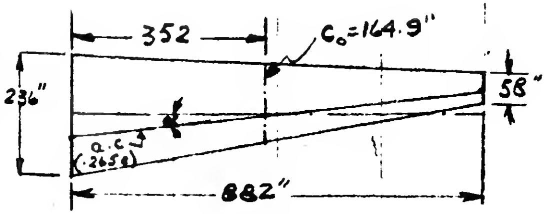
$\bar{c} = \frac{(S/2)}{(\frac{b}{2})} = \frac{1}{3} \times \frac{388910}{882} = 147\ in.$

$\frac{b}{2} = 882\ in.$

...IDATED VULTEE AIRCRAFT CORPORATION
SAN DIEGO DIVISION — SAN DIEGO, CALIFORNIA

SPANWISE PITCHING-MOMENT DISTRIBUTION
REVISED AIRFOILS & WING
ELEV. 0°

1	21	22	23	24	25	26	27	28	29	30
$y/\frac{b}{2}$	$C_{m_{a.c.}}$	$C_{m_{a.c.}}C$	X	$C_{l_b}X$	$C_{l_b}X+C_{m_{a.c.}}C$	$\int(25)c\,dy \times 10^{-5}$	C_{l_1} $(C_{l_a}+C_{l_b})$	$C_{l_1}X$	$C_{l_1}X+C_{m_{a.c.}}C$	$\int(29)c\,dy \times 10^{-5}$
		(21)×(2)		(20)×(23)	(22)+(24)	(25)×(5) ×10⁻⁵	(20)+(14)	(23)×(27)	(22)+(28)	(29)×(5) ×10⁻⁵
0	.0065	1.535	52.190	2.560	4.095	.850	.945	49.40	50.935	10.600
.100	.0083	1.710	39.110	1.270	3.180	2.450	.962	37.60	39.410	30.400
.200	.0105	2.100	26.020	.495	2.595	.918	.978	25.40	27.500	9.720
.300	.0120	2.380	12.740	.052	2.432	1.568	.994	12.90	15.280	9.850
.400	.0162	2.670	-.119	.001	2.671	.779	1.009	-.12	2.550	.743
.500	.0177	2.900	-12.196	.290	3.190	1.654	1.029	-13.50	-10.600	-5.500
.600	.0208	2.680	-26.270	.656	3.336	.279	1.055	-27.55	-24.870	-2.825
.600	.0208	2.680	-26.270	.656	3.336	.189	1.055	-27.55	-24.870	-1.413
.650	.0215	2.580	-32.810	.854	3.434	.728	1.067	-35.00	-32.420	-6.880
.700	.0225	2.480	-39.351	1.102	3.582	.350	1.077	-42.30	-39.820	-3.900
.750	.0230	2.345	-45.890	1.375	3.720	.670	1.080	-49.60	-47.255	-8.500
.800	.0242	2.280	-52.430	1.680	3.960	.328	1.063	-55.70	-53.420	-4.420
.850	.0255	2.165	-58.970	2.000	4.165	.625	1.047	-61.75	-59.585	-8.940
.900	.0273	2.075	-65.510	2.100	4.175	.140	1.003	-65.75	-63.675	-2.230
.900	.0273	2.075	-65.510	2.100	4.175	.070	1.003	-65.75	-63.675	-1.070
.925	.0282	2.020	-68.775	2.130	4.150	.262	.969	-66.50	-64.480	-4.060
.950	.0294	1.970	-72.044	2.230	4.200	.124	.900	-64.80	-62.830	-1.860
.975	.0308	1.925	-75.314	2.260	4.185	.231	.806	-60.65	-58.725	-3.240
1.000	.0325	1.885	-78.580	1.960	3.845	.049	.475	-37.40	-35.515	-.454
						Σ12.364				+61.313
										-55.292
										Σ+6.021



$\tan\alpha = .14827$

$$X = (352 - y)\tan\alpha$$

$$C_{m_{a.c.}} = \frac{\int (C_{l_1}X + C_{m_{a.c.}}C)\,c\,dy}{(\frac{S}{2})\,C_o} \quad (\text{FOR } C_L = 1.0)$$

$$C_{m_{a.c.}} = \frac{\int (C_{l_b}X + C_{m_{a.c.}}C)\,c\,dy}{(S/2)\,C_o} \quad (\text{FOR } C_L = 0)$$

$$S/2\ C_o = (5) \times 164.9 = 640.5$$

$$C_{m_{ac}} = \frac{12.364}{640.5} = +.0193 \quad @\ C_L = 0$$

$$C_{m_{ac}} = \frac{6.021}{640.5} = +.0094 \quad @\ C_L = 1.0$$

SPAN-LOAD DISTRIBUTION
REVISED AIRFOILS & WING
ELEV. -10°

2-ENG. TAILLESS DESIGN

1 $y/\frac{b}{2}$	2 MULT. x 3	3 CHORD C (in.)	4 ΔY in.	5 $\int c\,dy$ (2)x(3)x(4)	6 ε WASHOUT (DEG.)	7 α_{l_0}	8 β $(\varepsilon-\alpha_{l_0})$ (6)-(7)	9 $\int \beta c\,dy$ (8)x(5)	10 $\beta+\alpha_{R_0}$	11 C_{l_b} (faired)
0	1	236	44.1	10400	0	4.40	-4.40	-45200	-.78	-.047
.05	4	227		40000	-.15	4.37	-4.52	-181000	-.90	-.054
.10	2	218		19250	-.30	4.33	-4.63	-89000	-1.01	-.060
.15	4	209		36900	-.45	4.30	-4.75	-175500	-1.13	-.069
.20	2	200		17650	-.64	4.25	-4.89	-86300	-1.29	-.077
.25	4	191		33700	-.83	4.18	-5.01	-169000	-1.39	-.083
.30	2	183		16150	-1.04	4.10	-5.14	-83000	-1.52	-.091
.35	4	174		30700	-1.26	4.00	-5.26	-161500	-1.64	-.098
.40	2	165		14580	-1.54	3.87	-5.41	-80400	-1.79	-.107
.45	4	156		27500	-1.83	3.70	-5.53	-152000	-1.91	-.090
.50	1	147		6480	-2.04	3.54	-5.58	-36150	-1.96	.0
.50	1	147		6480	-2.04	-1.06	-.98	-6350	+2.64	.0
.55	4	138		24300	-2.12	-1.10	-1.02	-24800	2.60	.125
.60	2	129		11400	-2.20	-1.17	-1.03	-11750	2.59	.155
.65	4	120		20200	-2.30	-1.25	-1.05	-21200	2.57	.154
.70	2	111		9800	-2.42	-1.34	-1.08	-10600	2.54	.152
.75	4	102		18000	-2.56	-1.45	-1.11	-20000	2.51	.150
.80	2	94		8300	-2.74	-1.60	-1.14	-9470	2.48	.149
.85	4	85		15000	-2.95	-1.78	-1.17	-17550	2.45	.147
.90	2	76		6720	-3.20	-2.06	-1.14	-7670	2.48	.149
.95	4	67		11800	-3.55	-2.43	-1.12	-13200	2.50	.150
1.00	1	58		2560	-4.00	-2.98	-1.02	-2610	2.60	.154
				Σ 387870				Σ -1404850		

$$\alpha_{R_0} = -\left(\frac{-1404850}{387870}\right) = 3.62^\circ$$

TABLE IV

ESTIMATION OF STALL CHARACTERISTICS
REVISED WING
2-ENG. TAILLESS DESIGN

1	2	3	4	5	6	7	8	9
$y/\frac{b}{2}$	$C_{l_{a_1}}$	C_{l_b}	$C_{l_{a1.5}}$	$C_{L_{1.5}}$	$y/\frac{b}{2}$	C_{l_b} (faired)	$C_{l_{a1.35}}$	$C_{L_{1.35}}$
		$e=0°$				$e=-10°$		
0	.897	.049	1.348	1.397	0	-.047	1.210	1.163
.100	.927	.035	1.392	1.427	.10	-.060	1.253	1.193
.200	.959	.019	1.440	1.459	.20	-.077	1.295	1.218
.300	.990	.004	1.485	1.489	.30	-.091	1.335	1.244
.400	1.021	-.012	1.533	1.521	.40	-.107	1.380	1.273
.500	1.051	-.022	1.580	1.558	.50	0	1.420	1.420
.600	1.080	-.025	1.620	1.595	.50	0	1.420	1.420
.600	1.080	-.025	1.620	1.595	.55	.125	1.441	1.566
.650	1.093	-.026	1.643	1.617	.60	.155	1.460	1.615
.700	1.105	-.028	1.660	1.632	.65	.154	1.477	1.631
.750	1.110	-.030	1.665	1.635	.70	.152	1.495	1.647
.800	1.095	-.032	1.642	1.610	.75	.150	1.500	1.650
.850	1.081	-.034	1.625	1.591	.80	.149	1.480	1.629
.900	1.035	-.032	1.552	1.520	.85	.147	1.464	1.611
.900	1.035	-.032	1.552	1.520	.90	.149	1.399	1.548
.925	1.000	-.031	1.500	1.469	.95	.150	1.259	1.409
.950	.931	-.031	1.400	1.369	1.00	.154	.675	.829
.975	.836	-.030	1.253	1.223				
1.000	.500	-.025	.750	.725				

Defendants' Exhibit A—(Continued)

Consolidated Vultee Aircraft Corporation
San Diego Division

Model......Airplane Report No. ZA-101, App. D

Appendix D

Method for the Calculation of the Leading Edge Radius of an Airfoil

ETHOD FOR THE CALCULATION OF THE LEADING EDGE RADIUS OF AN AIRFOIL

(Reference 8)

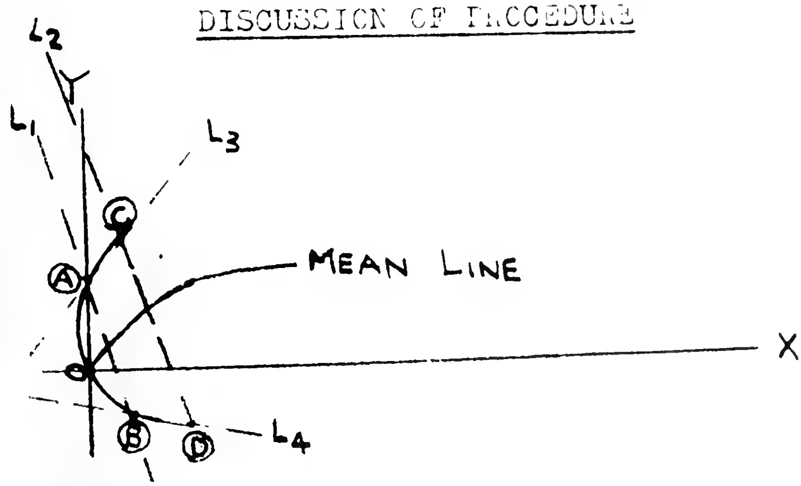
SUMMARY

The following method is used for the calculation of the ing edge radius of an airfoil. The slope of the camber-line at on (0,0) shown in Appendix B is estimated in agreement with the of the camber line at station (0,0) as calculated by the pro- e outlined below.

Leading edge radii and mean camber-line slopes at station (0,0) he proposed revised wing airfoil section are:

	Radius	Slope
Root	3.7532 $\frac{c}{100}$	.1298
48% Span	2.3667 $\frac{c}{100}$	.3611
Tip	1.7159 $\frac{c}{100}$	.5961

DISCUSSION OF PROCEDURE



Select the four points (A) (B) (C) (D) which are nearest to e leading edge and of which the X and Y coordinates are known.

1068

Then find the equation of the conic section through the four points and the origin. The radius of osculating circle at the origin is taken as the leading edge radius.

Notation m = slope

b = y intercept

$L = y - mx - b = 0$ is the equation of a straight line

Subscripts A, B, C, D, O refer to the points A, B, C, D, O.

Subscript 1 refers to a line AB passing through A & B.

Subscript 2 refers to a line CD.

Subscript 3 refers to a line AC.

Subscript 4 refers to a line BD.

Then (1) $m_1 = \dfrac{Y_A - Y_B}{X_A - X_B}$

(2) $b_1 = Y_A - m_1 X_A = Y_B - m_1 X_B$

Similarly m_2, m_3, m_4, b_2, b_3 & b_4 are found.

Using the degenerate form of the general conic as the sum of products of two linear equations.

(3) $(L_1) \times (L_2) + k (L_3) \times (L_4) = 0$

(4) $(y - m_1 x - b_1)(y - m_2 x - b_2) + k(y - m_3 x - b_3)(y - m_4 x - b_4) = 0$

Imposing the condition that the conic has to pass through point O ($x = 0$, $y = 0$) and solving for k gives

(5)
$$k = -\frac{b_1 b_2}{b_3 b_4}$$

Taking x-derivatives of (4) yields

(6)
$$(y - m_1 x - b_1)\left(\frac{dy}{dx} - m_2\right) + \left(\frac{dy}{dx} - m_1\right)(y - m_2 x - b_2) + k\left[(y - m_3 x - b_3)\left(\frac{dy}{dx} - m_4\right) + \left(\frac{dy}{dx} - m_3\right)(y - m_4 x - b_4)\right] = 0$$

Setting x = 0, y = 0 and solving for $\frac{dy}{dx}$ yields the slope of conic at the origin

$$m_o = + \frac{b_1 m_2 + b_2 m_1 + k(b_3 m_4 + b_4 m_3)}{b_1 + b_2 + k(b_3 + b_4)}$$

Again taking x-derivatives of (6) yields

$$(y-m_1 x-b_1) \frac{d^2y}{dx^2} + \left(\frac{dy}{dx} - m_1\right)\left(\frac{dy}{dx} - m_2\right)$$

$$+ \left(\frac{dy}{dx} - m_1\right)\left(\frac{dy}{dx} - m_2\right) + \frac{d^2y}{dx^2} (y-m_2x-b_2)$$

$$+ k\left[(y-m_3x-b_3) \frac{d^2y}{dx^2} + 2\left(\frac{dy}{dx} - m_3\right)\left(\frac{dy}{dx} - m_4\right)\right.$$

$$\left. + \frac{d^2y}{dx^2} (y-m_4x - b_4)\right] = 0$$

ing for $\frac{d^2y}{dx^2}$ and substituting x = 0; y = 0 gives

$$\frac{d^2y}{dx^2} = \frac{2\left[(1 + k) m_o^2 - \left\{m_1 + m_2 + k (m_3 + m_4)\right\} m_o + m_1 m_2 + k m_3 m_4\right]}{b_1 + b_2 + k (b_3 + b_4)}$$

It is now possible to use the general equation for the radius he osculating circle for point 0

$$(10) \quad R_o = \frac{\left[1 + m_o^2\right]^{3/2}}{\left(\frac{d^2y}{dx^2}\right)_o}$$

The coordinates of the center of the circle of radius R are

BY
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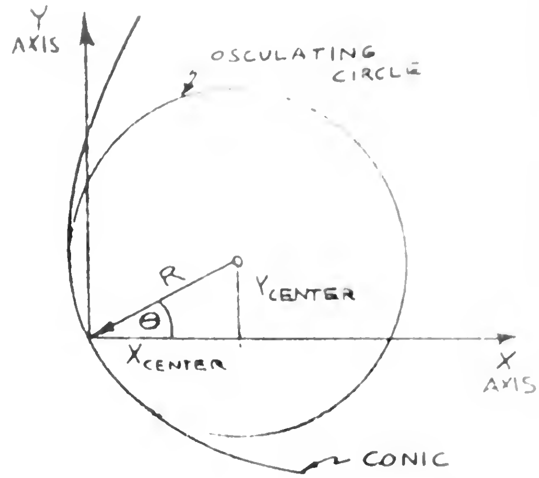
APPROVED



11 $X_{center} = R_0 \cos \theta$

12 $Y_{center} = R_0 \sin \theta$

The value of θ may be found, where:

(1) $\cot \theta = -m_0$ or if no tables of functions are available

(1) $\cos \theta = -m_0 / \sqrt{1+m_0^2}$

(1) $\sin \theta = 1 / \sqrt{1+m_0^2}$

Admitted November 22, 1950.

DEFENDANTS' EXHIBIT EE

The Glenn L. Martin Company Model B-26
Baltimore, Maryland Page No. 1

G.L.M. Engineering Report No. 1326

Wind Tunnel Investigation of the B-26
Stall Characteristics

Engineering Report No. 1326

The Glenn L. Martin Company
Baltimore, Maryland

July 19, 1940

Prepared by: A. J. Trimble, Jr.

Checked by: E. B. Schaefer.

Approved by: V. Outman,
Chief of Aerodynamics.

Approved by: Paul E. Hovgard,
Chief Research Engineer.

Defendants' Exhibit EE—(Continued)

The Glenn L. Martin Company — Model B-26
Baltimore, Maryland — Page No. 2

G. L. M. Eng. Rep. No. 1326

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The Glenn L. Martin Company Model B-26
Baltimore, Maryland Page No. 3

G.L.M. Eng. Rep. No. 1326

Wind Tunnel Investigation of B-26
Stall Characteristics

After a reconsideration of the probable stalling characteristics on the Model B-26 (Glenn L. Martin Model 179) it was decided that instead of waiting until the airplane is flown to see if tip stall occurs, a change should be made to diminish the possibility of poor behavior at the stall. An extensive test program was conducted at the Massachusetts Institute of Technology, Wright Brothers' Wind Tunnel, to determine the steps to be taken and the results are reported herein.

The scope of the investigation, and necessarily this report, was limited to those physical changes deemed advisable on the actual airplane in order not to delay delivery. Change in wing profile shape has been confined, therefore, to an area forward of the 10% chord line and outboard of Station 255 to the tip. In addition, the use of spoilers was also considered a possibility in the event that other methods failed to produce the correct effects.

With these limitations in mind, it appears that Leading Edge No. 2, illustrated on page 11 produces the desired effect most efficiently. In the following report, the justification of this choice will be brought forth by first, a short discussion of the basic

Defendants' Exhibit EE—(Continued)

problems of wing stall; second, a description of the model and tests; and third, a presentation and discussion of the data.

Basic Considerations

The criterion for desired stalling characteristics of an airplane must first be agreed upon, after which several methods for obtaining these characteristics are open to the designer along with methods for analyzing and predicting the results. In the

case of the B-26, the field of possible wing design in this stage of the airplane's construction is limited because of a time consideration.

Desired Stalling Characteristics

The stall should start at the wing root to produce the most desirable and safest effect. Such a condition will result in a reduced downwash at the tail causing a diving moment tending to prevent the pilot from increasing the angle of attack and stalling the tips. Tail buffeting, a result of the turbulent air from the stalled root sections, warns the pilot that he has reached a stalled condition. A midpanel stall between the nacelle and inboard end of the aileron causes neither serious tail buffeting nor a diving moment. The desirability of completely eliminating tip stall is universally recognized.

Defendants' Exhibit EE—(Continued)

Design Methods and Limitations

The desired stall may be regulated; first, by warping the wing either geometrically or aerodynamically; second, varying plan form shape, i.e., taper ratio; third, varying thickness ration along the span; fourth, using slots to delay the stall; fifth, using spoilers to cause stall.

The section of wing available for design change to assure root stall is illustrated on page 10. These limitations narrow the field of design methods to a leading edge change which might incorporate a slot, or a drooped nose effectively warping the tip of the wing. There is no design limitation on the use of spoilers.

In addition to physical limitations, further restrictions are present because the root stall must be obtained with the least possible increase in drag and the greatest possible increase in maximum lift.

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Methods of Stall Analysis

The stall of the B-26 wing has been analyzed both at full scale Reynold's Number and Model Reynold's Number in accordance with the method set forth in NACA Technical Report No. 572. The results appear on pages 12 and 13. Approximate values of $C_{L\,max}$ and ΔC_L 's due to varying Reynold's Number were estimated from the data available on

Defendants' Exhibit EE—(Continued)

the standard symmetrical airfoil series, (NACA OO—) with the maximum thickness at the 30% chord station. Since the Model B-26 wing contours are those of the NACA OO—64 airfoils with the maximum thickness at the 40% chord station, it is very likely that some discrepancy may exist in these stall diagrams.

The knowledge that the model did not stall exactly as indicated by the stall diagram, but nearer the tip, led to an investigation of the possibilities of correcting this condition, for the same discrepancy might exist on the full scale airplane. Such a condition would be aggravated by propeller wash. For these reasons, a study of the changes possible on the airplane and subsequently a complete test program of the various corrective possibilities has been undertaken.

Method of Test

The test program was conducted in the Massachusetts Institute of Technology Wright Brothers' Wind Tunnel with an ⅛ scale model of the B-29 which conformed in all respects to the airplane as being built.

Apparatus

The model used in this program was identical with the one used in previously reported tests (E.R. 1308). The leading edge of both wings was cut out as shown on page 10 to accommodate various

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Baltimore, Maryland Page 6

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leading edges which are illustrated on page 11. In addition the old B-26 model with the twisted wing (2° washout) was also tested. Flaps, airflow and Block Nacelles were available for the model.

The wind tunnel is equipped with a grid which raises the normal turbulence factor of 1.015 to 2.5. The speeds used were 125 M.P.H. with the grid and 150 M.P.H. without the grid. The accuracy obtainable in coefficient form for this model is:

$$C_D = .0002$$
$$C_L = .002$$

Procedure

To determine stall characteristics, pictures of tufts were taken at various angles of attack with the different leading edges at the same time lift data were taken. Tufts have no effect on lift in this particular case (Page 14). For these stall runs, the grid was used in the tunnel, and the model was equipped with airflow nacelles and deflected flaps (55°) because this configuration results in the most undesirable stall pattern, and is most likely to agree with flight conditions. The tail was not on the model during these runs because of the likelihood of severe buffeting. The airspeed was 125 M.P.H.

For drag tests, the model was equipped with block

Defendants' Exhibit EE—(Continued)

nacelles, flaps zero, and tail in place. No grid was used in the tunnel and the airspeed was 150 M.P.H.

In addition to these tests, unsymmetrical stalls were investigated.

Discussion

The lift and drag results and stall pictures are self-explanatory for the most part, but the choice of the best compromise is not quite as apparent from these data as it might be. The justification of the final choice is discussed in the following section.

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G.L.M.Eng. Rep. No. 1326

Results

Pages 15 to 22 illustrate the stall patterns and lifts developed by the various leading edges. For the most part, the stalls were symmetrical. Occasionally a root stall occurred on the right side before the left wing had stalled. In the case of the No. 2 leading edge, as shown on page 16, the right wing stalled first at the aileron. The left wing could not be stalled. Close inspection showed slightly more camber in the left side than in the right. Premature stalling of one wing delays the stall of the other wing. By deliberately stalling the left wing with spoilers along the leading edge, the right wing was found to stall 1 to 1½ degrees later than with the left wing unstalled. Hence, an apparent difference in stall commencement of 2 degrees between

Defendants' Exhibit EE—(Continued)

the right and left wing may be brought about by a ½ degree discrepancy in wing contours or stream rotation. To correct for this condition, the right wing with the No. 2 leading edge was mudded to attempt to develop a symmetrical stall. The $C_{L\,max}$ obtained is shown on page 24 and is considered the best estimate of the performance of this design.

A drag summary plot appears on page 25. On page 26 the drag of spoilers used to produce a root stall with leading edges numbers 1 and 2 are plotted.

Stall Characteristics and $C_{L\,max}$

Inspection of this data indicates that no leading edge satisfies the requirement that the stall start at the root. Reference to the stall characteristics plots on pages 12 and 13 shows that at full scale Reynold's Number, there should be a tendency for the root to stall relatively sooner with respect to the tip than at model scale Reynold's Number. Leading edges Nos. 3, 6, and 7 satisfactorily delay tip stall but create a stall in the mid-panel, an unsatisfactory condition, and it very likely that scale effect will not be great enough to transfer this mid-

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panel stall to the root. In other words, a leading edge design permitting the tip sections to reach too high a lift coefficient when the wing sections adjacent to this leading edge change are approaching

Defendants' Exhibit EE—(Continued)

a stall condition, will hasten the stall in these unchanged mid-panel sections. Too great a delay in tip stall must be avoided, or drastic spoiling of the root section lift will be required, resulting in a very low overall maximum lift coefficient. Verification of this fact is apparent in the maximum lift comparisons of page 23, where the No. 7 leading edge affords a lower overall $C_{L\,max}$ than the No. 2 leading edge.

From the stall standpoint, the No. 2 leading edge is the best solution. $C_{L\,max}$ is increased .22, (page 24) indicating a delayed tip stall which is verified by the stall pictures. Scale effect enhances the use of this leading edge because the stall characteristic plots of pages 12 and 13 show an increased margin at the tip and decreased margin at the root; at the same time, leading edge No. 2 will not effect a large enough change in lift distribution to bring about a premature stall in the mid-panel. Admittedly, a root stall is not produced on the model by the use of this design, but scale effect will tend to change the stall characteristics of the model, moving the stall inboard to the root.

If this condition is not realized in flight, it may be obtained by placing small spoilers on the root section similar to those tested on the model (page 10). The change in lift caused by the spoilers is plotted on page 24. The actual lift and stall pictures are shown on page 21. Should the spoilers be needed in flight, the resulting airplane characteristics will be more acceptable than with any other leading edge design, for, as already pointed out, the other leading edges would produce a very definite mid-panel stall

Defendants' Exhibit EE—(Continued)

which could necessitate large spoilers on the root section to move the stall inboard. The result would be a very marked decrease in $C_{L\,max}$ and a great increase in drag. A root stall may be produced on the original wing with spoilers but the resultant $C_{L\,max}$ is extremely low in comparison with leading edge No. 2 (page 24). Comparisons of the original twisted B-26 wing and the more recent wing show very little difference in stall characteristics or $C_{L\,max}$ (pages 15 and 22).

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Drag

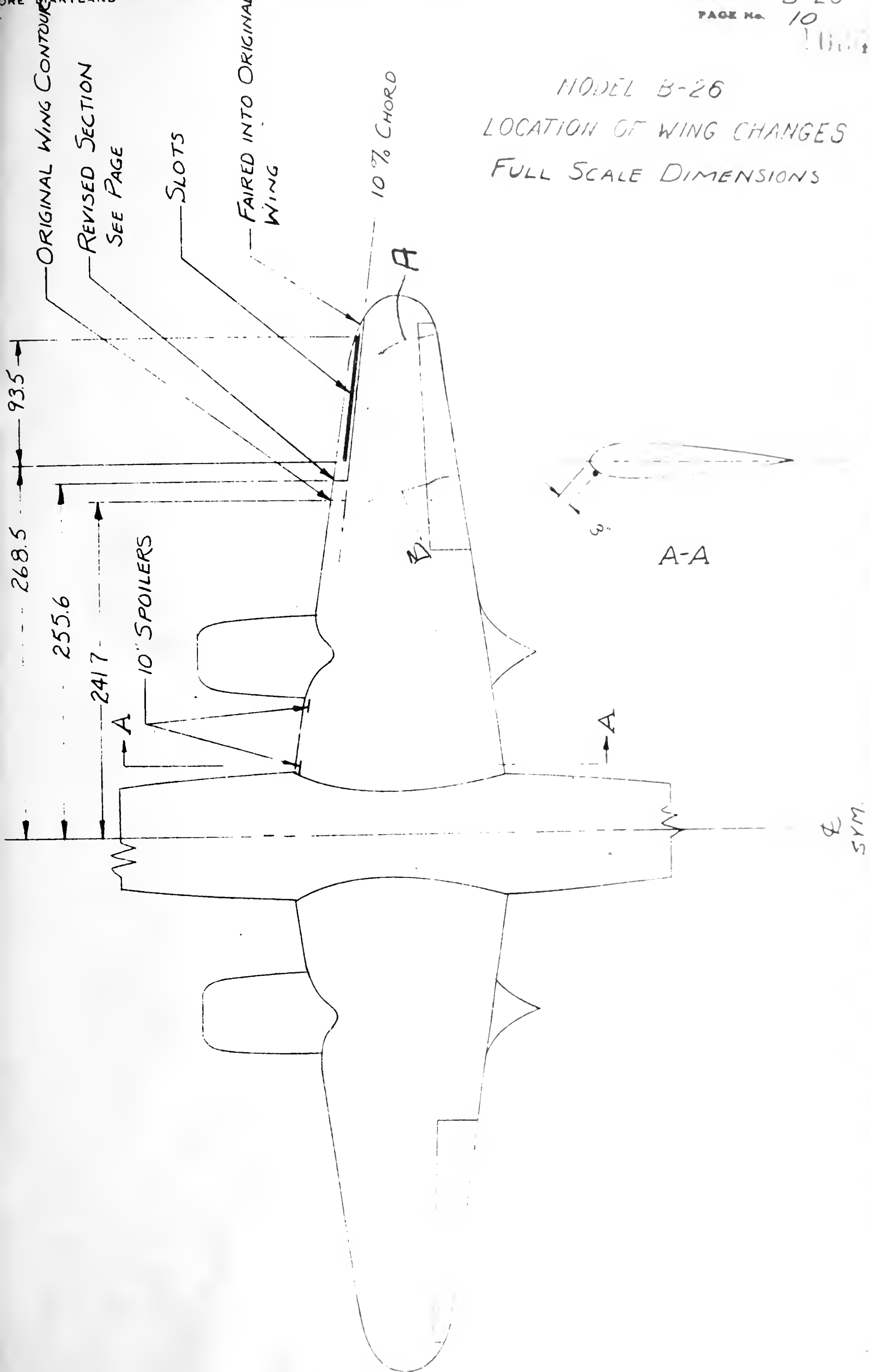
In addition to affording the best compromise in $C_{L\,max}$ and stall pattern, the No. 2 leading edge has less drag than any of the other configurations. Should the spoilers be found necessary, the drag is also slight. The comparisons of the No. 1 and No. 2 leading edge drags with spoilers is shown on page 26.

Conclusions

As a result of this investigation, the No. 2 leading edge is being incorporated in the design of the B-26 wing.

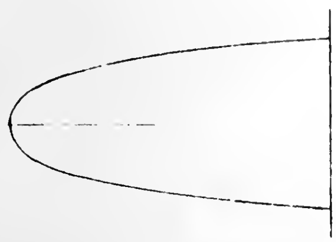
In case the stalling characteristics are not quite satisfactory in flight, it will be possible to completely correct it by adding a spoiler similar to those tested on the wind tunnel model.

MODEL B-26
LOCATION OF WING CHANGES
FULL SCALE DIMENSIONS

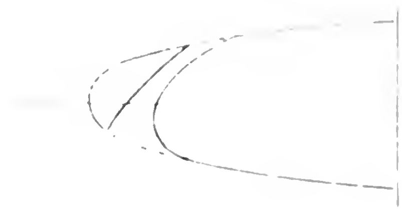


MODEL B-26
TYPICAL SECTIONS THROUGH LEADING EDGE AT STA. 350

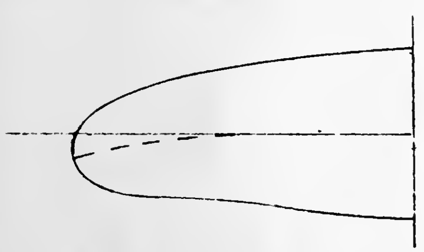
NOTE: ALL MODIFICATIONS FAIR INTO ORIGINAL CONTOUR AT 10 PERCENT CHORD STATION.



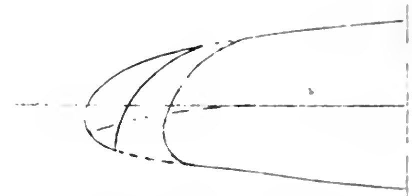
LEADING EDGE NO. 1
ORIGINAL CONTOUR
REF. DWG. W.T. 179-33



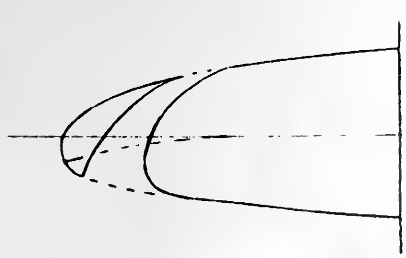
LEADING EDGE NO. 5
SAME AS CONTOUR NO 1
SLOT ADDED
REF. DWG. W.T. 179-96 5 OF 7



LEADING EDGE NO. 2
ORIGINAL CONTOUR MOD-
IFIED APPROX TO
220-- SERIES AIRFOIL
REF DWG W.T. 179-196 2 OF 7



LEADING EDGE NO. 6
CONTOUR NO 2 WITH SLOT
REF DWG W.T. 179-196 6 OF 7



LEADING EDGE NO. 3
SAME AS CONTOUR NO. 2
SLOT ADDED
REF. DWG W.T. 179-196 3 OF 7

LEADING EDGE NO. 7
ORIGINAL CONTOUR MODIFIED APP.
TO 420-- SERIES AIRFOIL
REF DWG W.T. 179-196 7 OF 7

Prepared By ___
Date ___ 40

THE GLENN L. MARTIN COMPANY
BALTIMORE, MD.

Page 12
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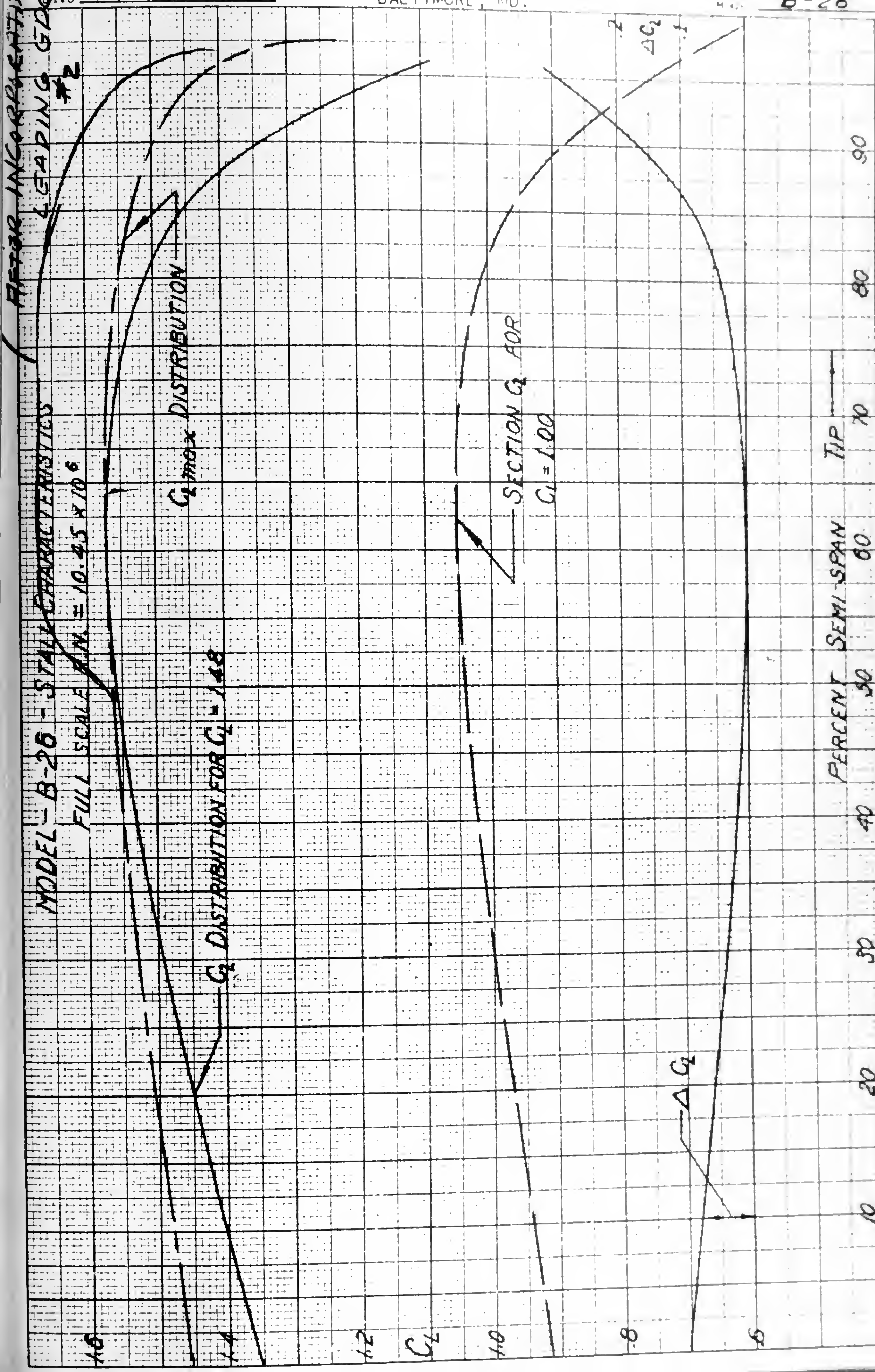
Model B-26



Prepared By L. Cooper
Date 7-22-40

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BALTIMORE, MD.

Page 13
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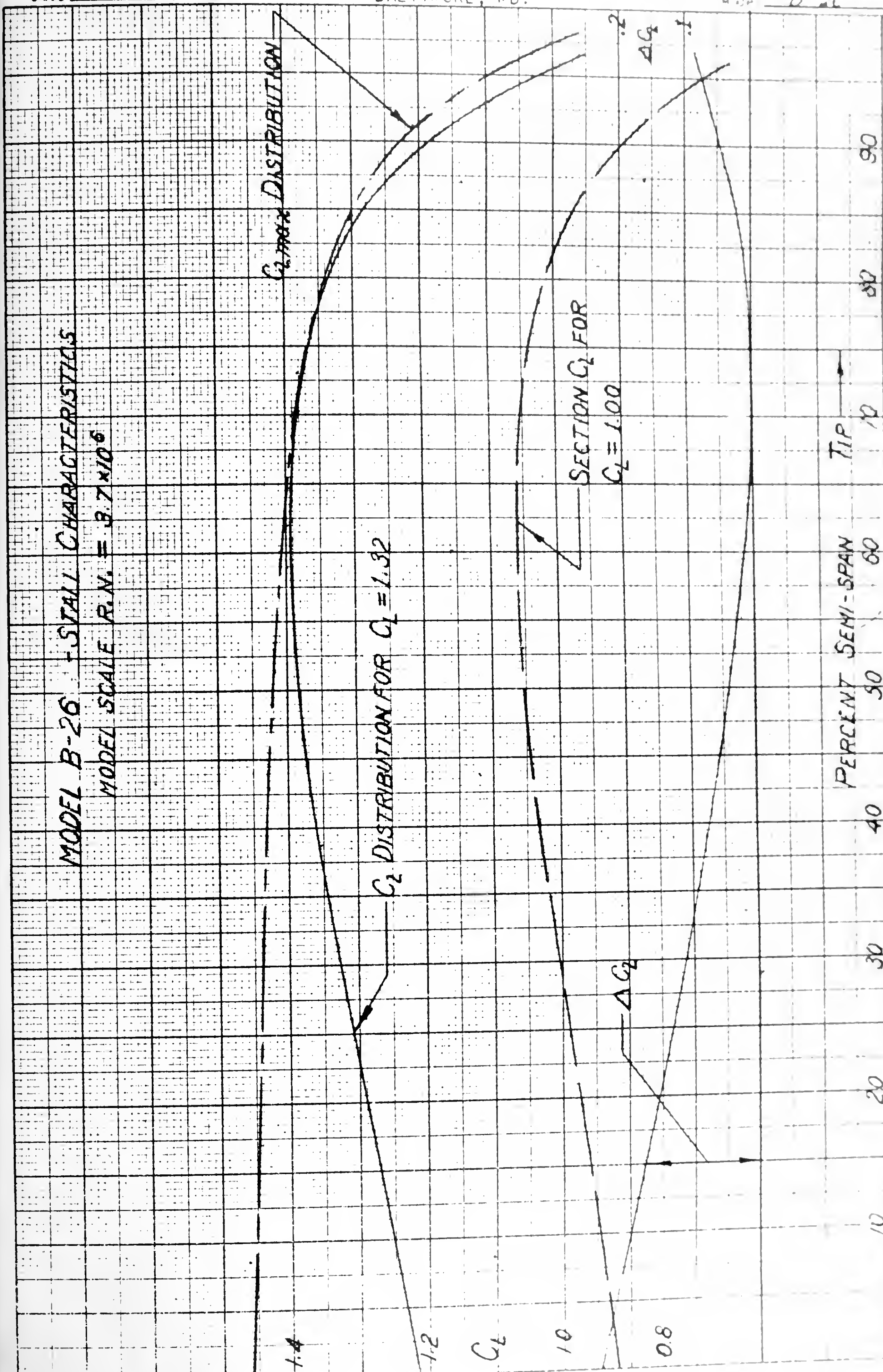

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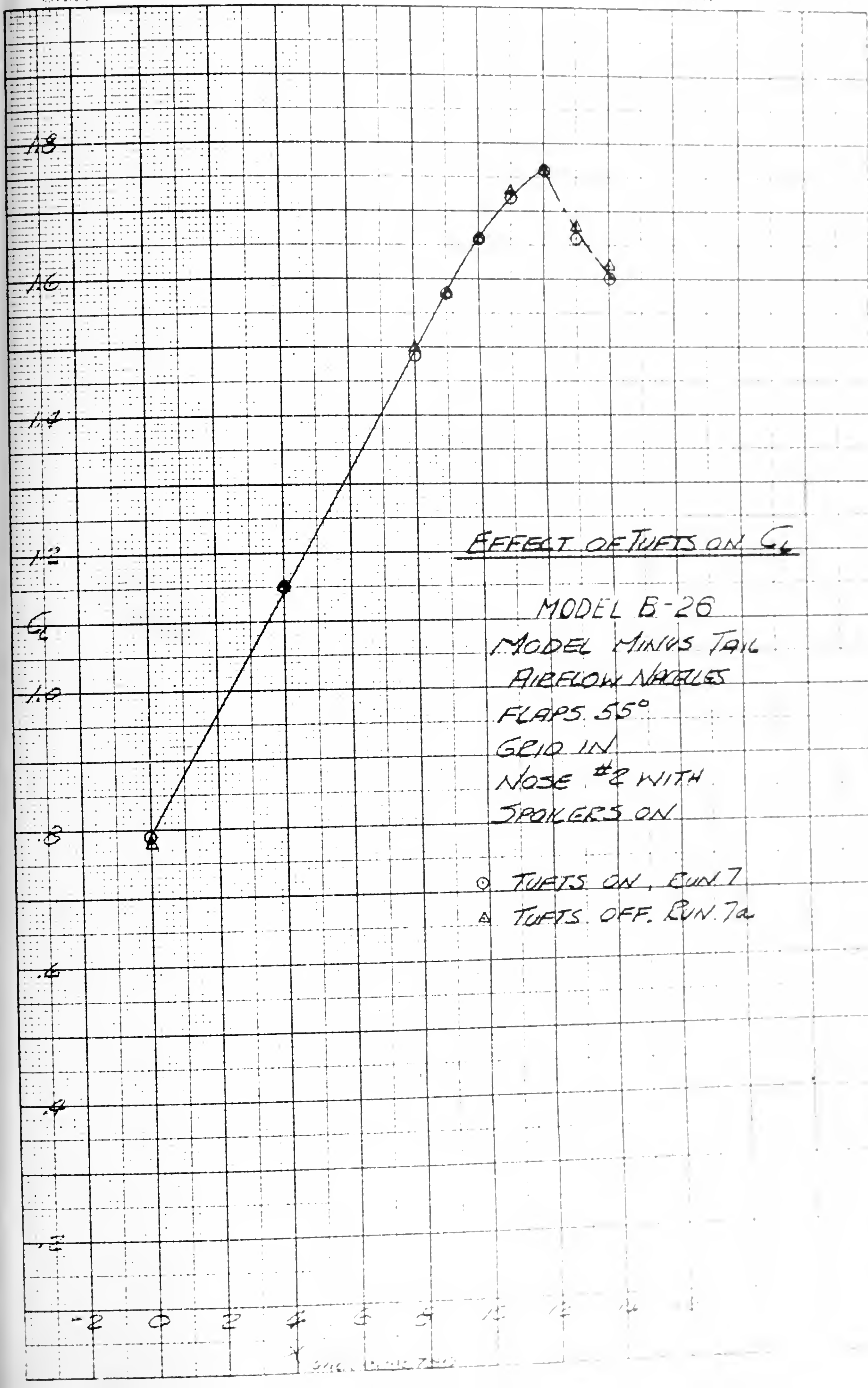
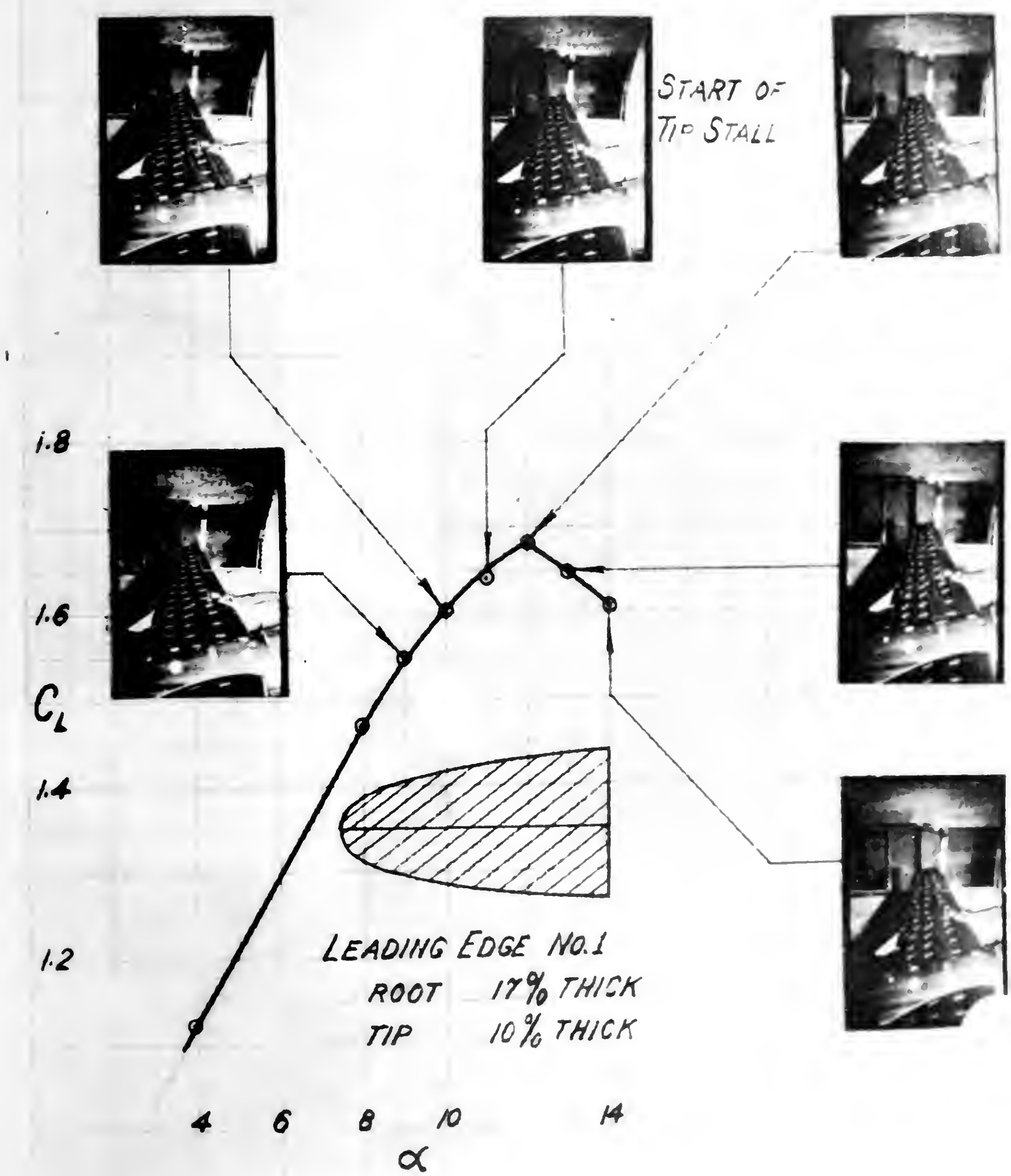


1.8
1.6
1.4
1.2
C_L
1.0
.8
.6
.4
.2
-2
0
2
4
6
8
10
12
14
EFFECT OF TUFTS ON C_L
MODEL B-26
MODEL MINUS TAIL
AIRFLOW NACELLES
FLAPS 55°
GRID IN
NOSE #2 WITH
SPOILERS ON
⊙ TUFTS ON, RUN 7
△ TUFTS OFF, RUN 7a

MODEL 179

C_L VS α AND STALL PATTERNS

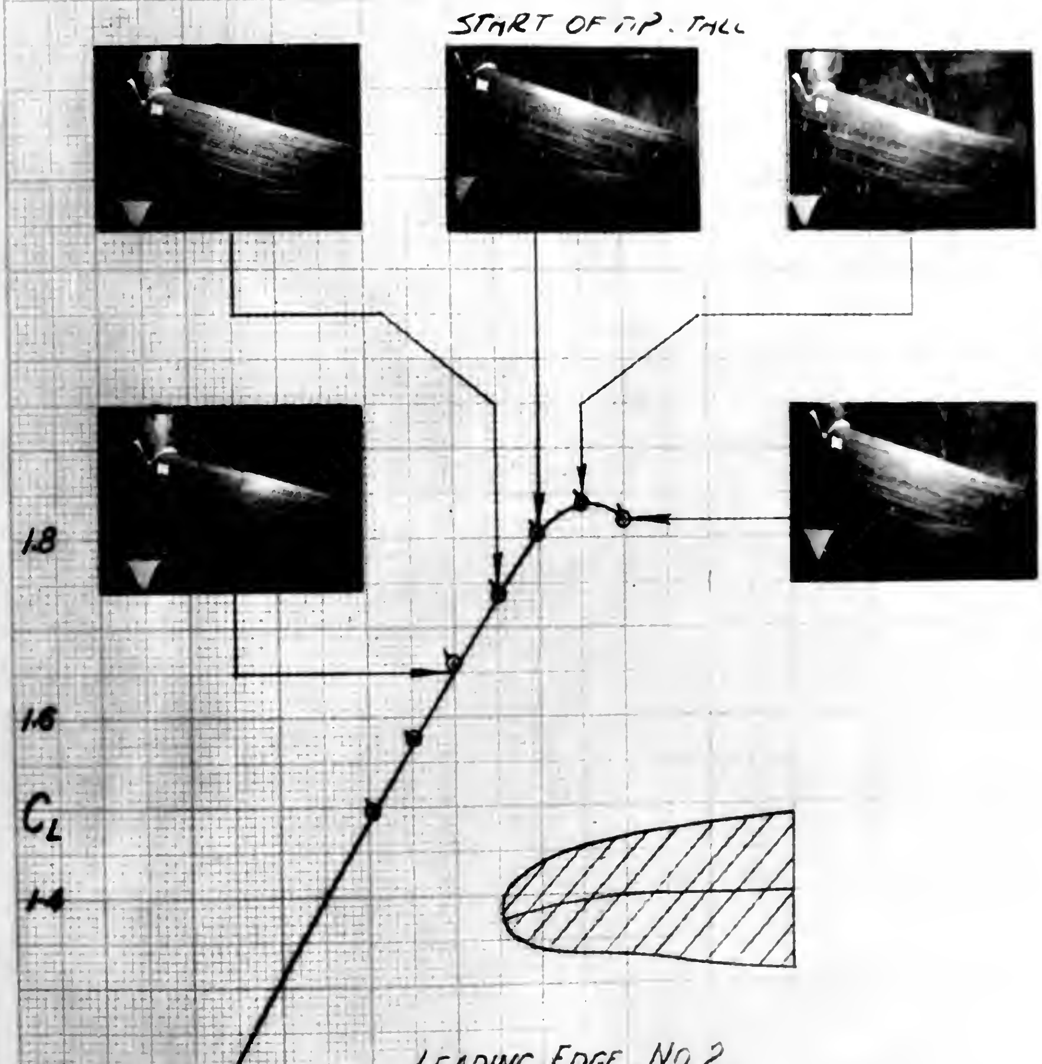
R.N. = 3.7×10^6
FLAPS 55°
AIRFLOW NACELLES



MODEL 179

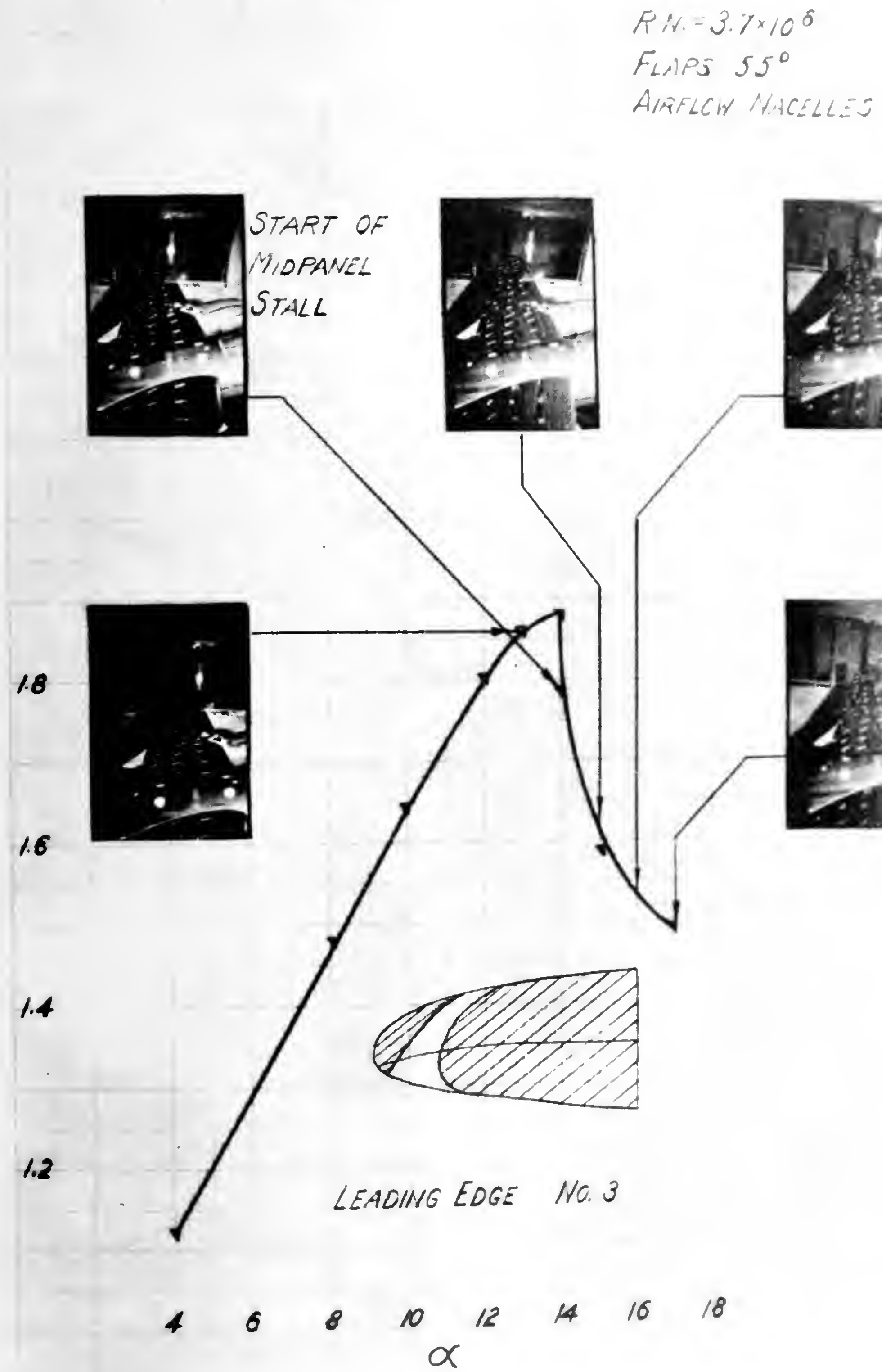
C_L VS α AND STALL PATTERNS

R.N. 3.7×10^6
FLAPS 55°
AIRFLOW NACELLES



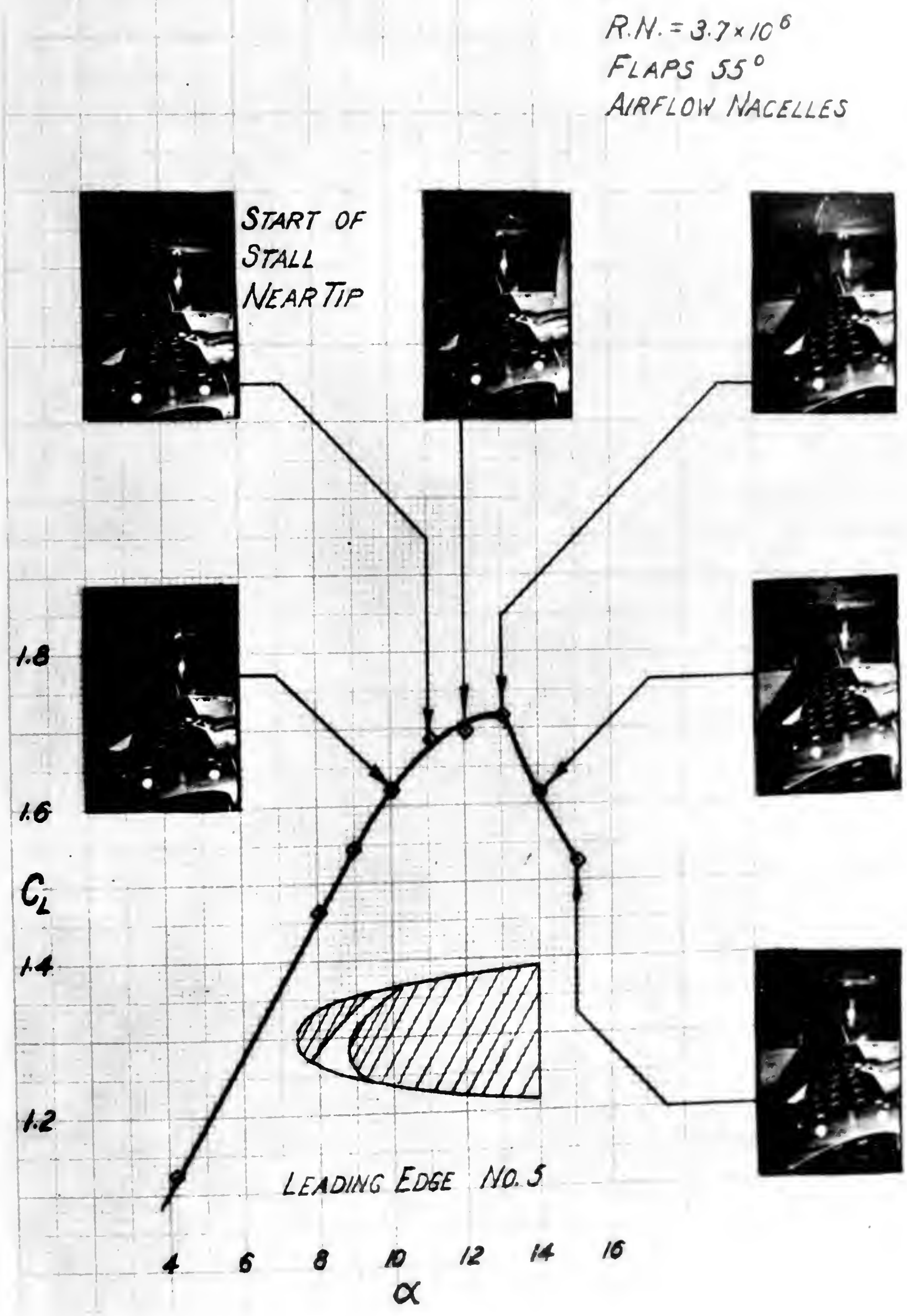
MODEL 179

C_L VS α AND STALL PATTERNS



MODEL 179

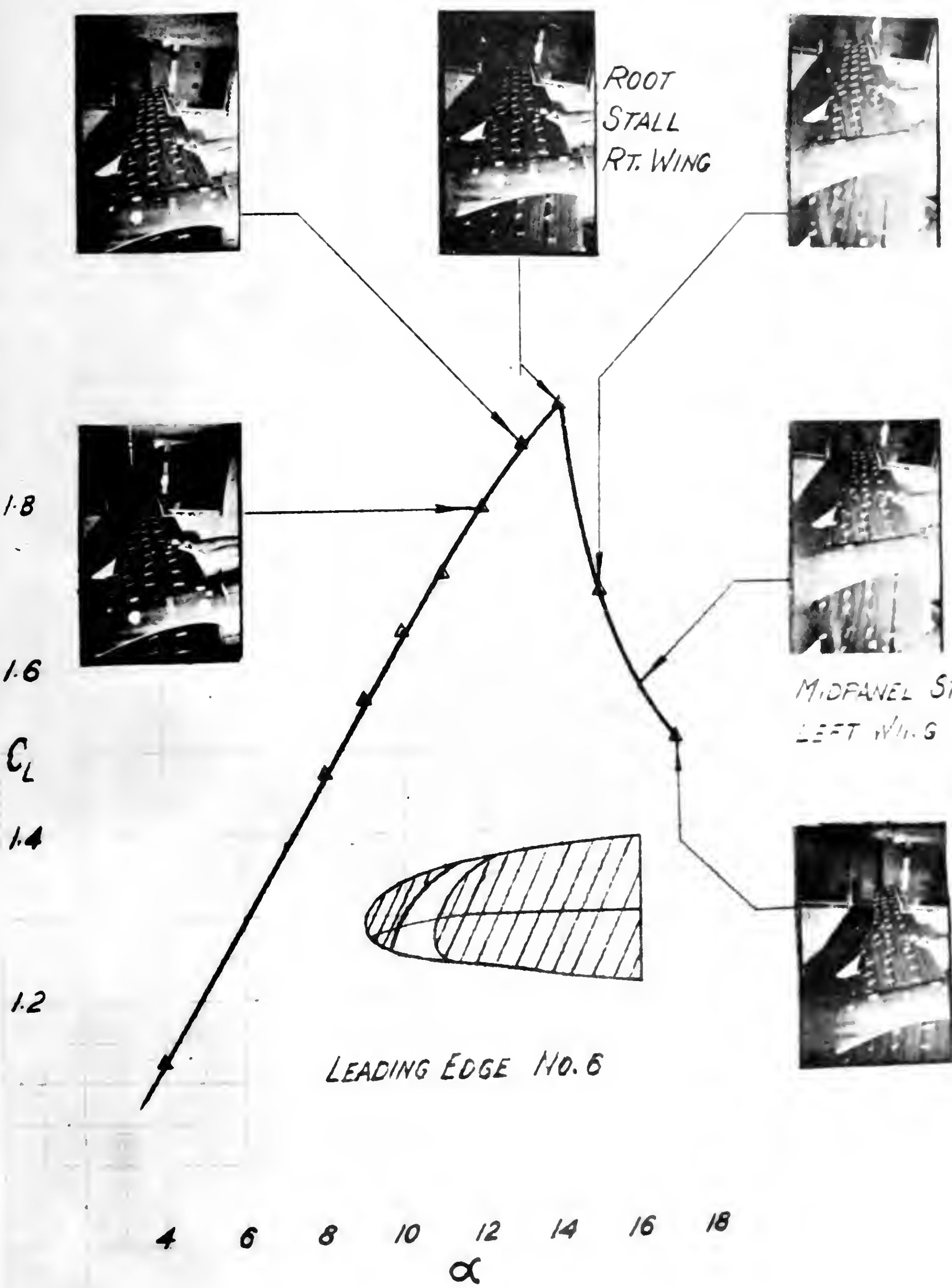
C_L VS α AND STALL PATTERNS



MODEL 179

C_L VS α AND STALL PATTERNS

RN = 3.7 × 10^6



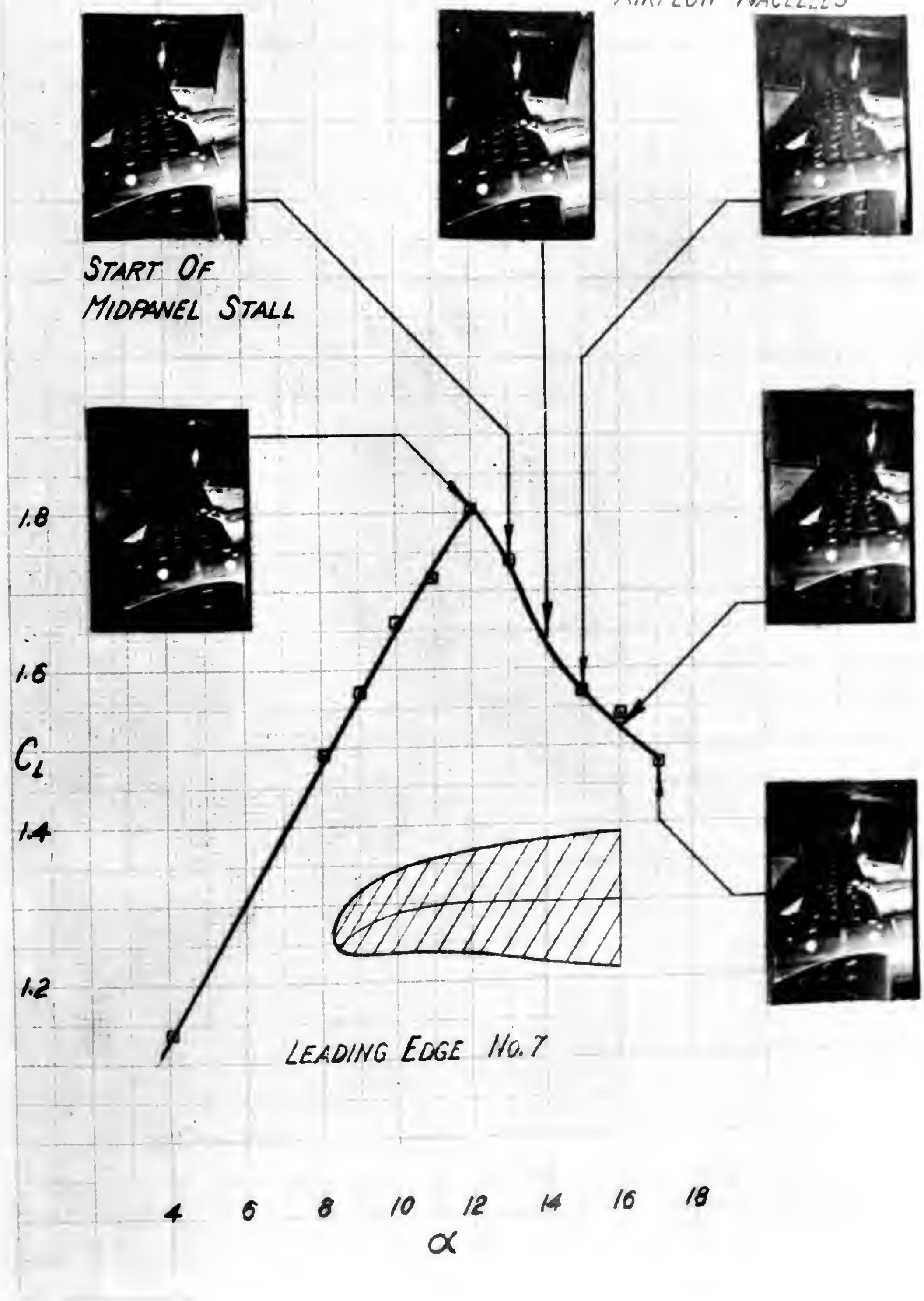
MODEL 179

C_L VS α AND STALL PATTERNS

R.N. = 3.7×10^6

FLAPS 55°

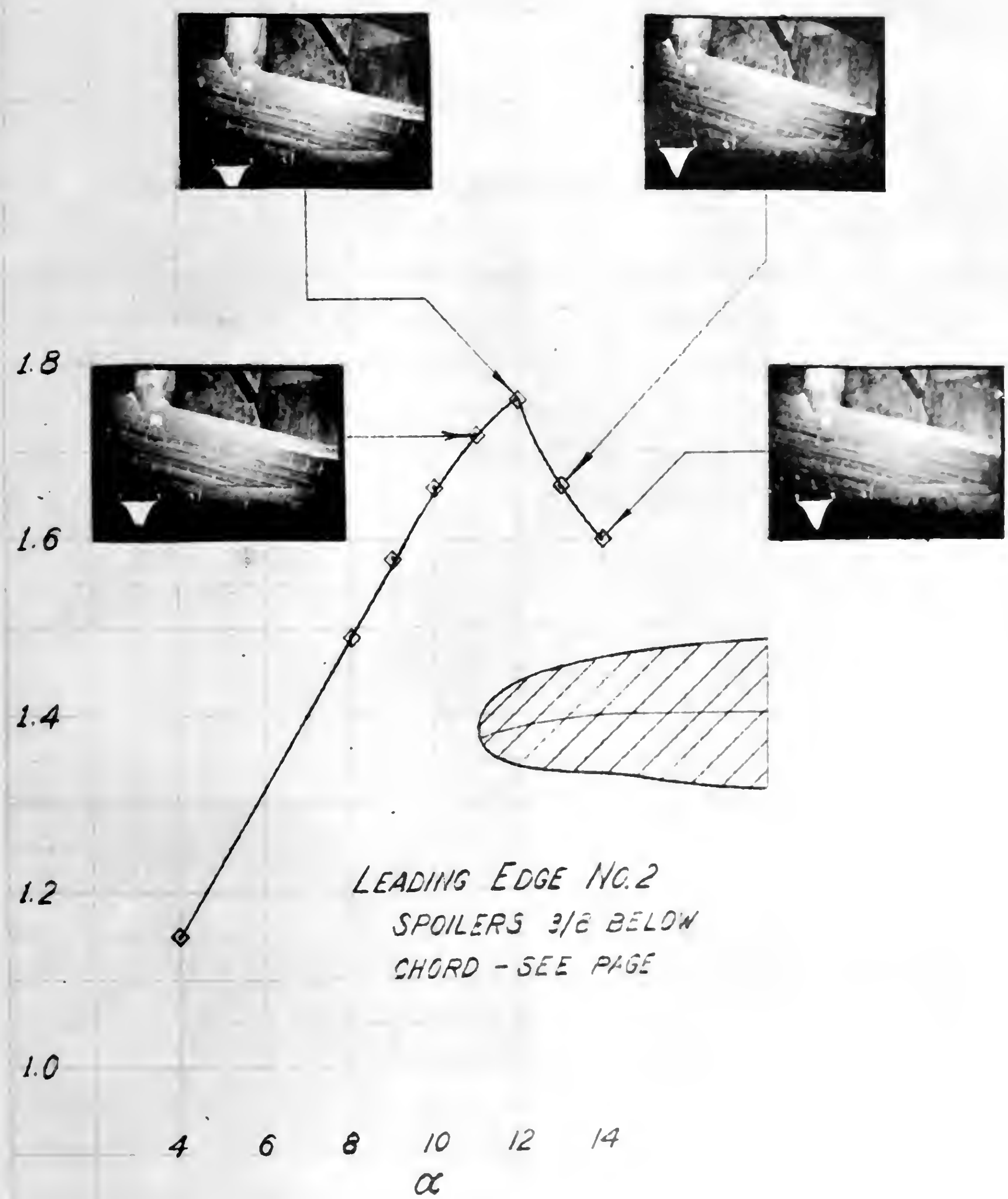
AIRFLOW NACELLES



MODEL 179

C_L VS α AND STALL PATTERNS

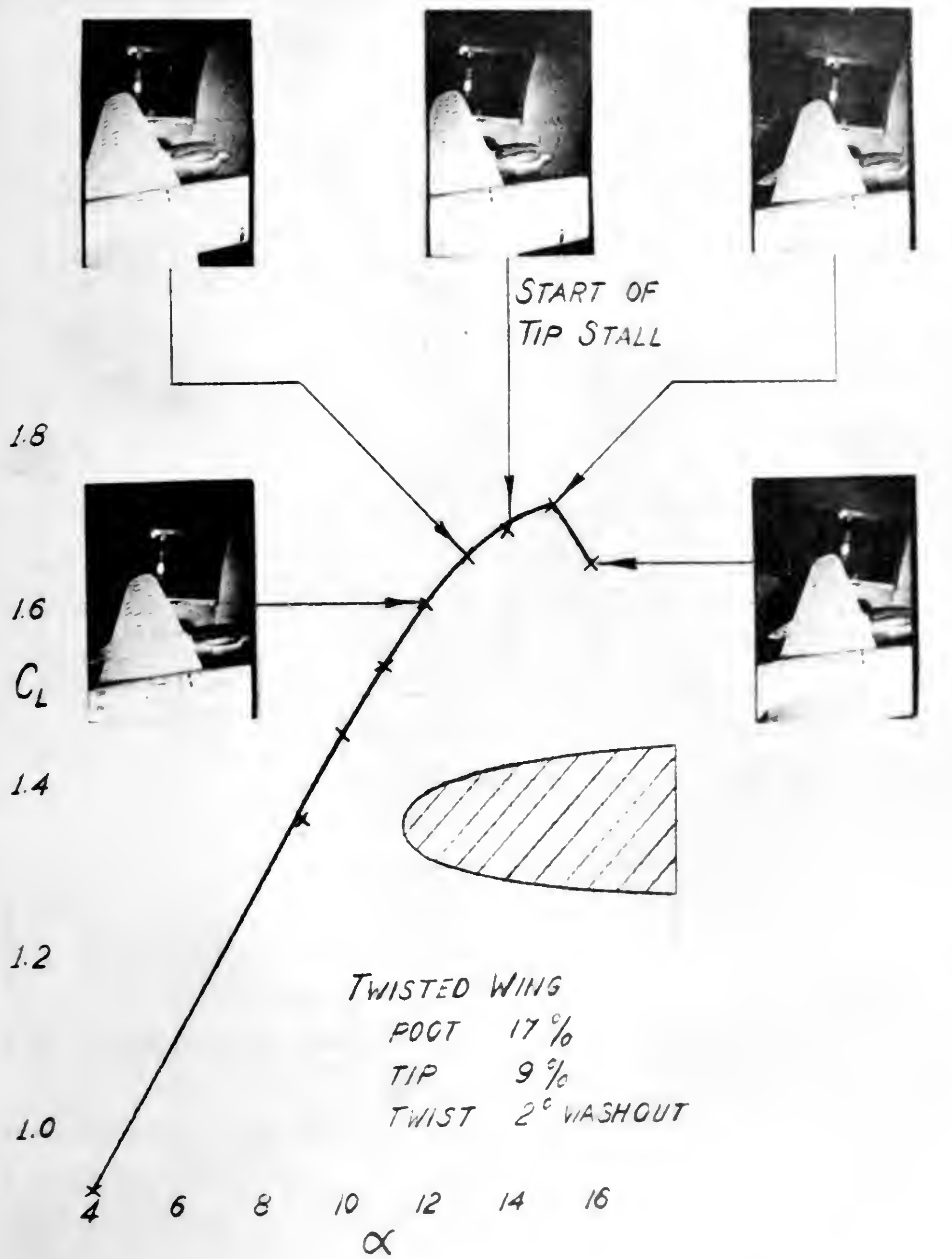
R.N. 3.7×10^6
FLAPS 55
AIRFLOW NACELLES



MODEL 179

C_L vs α AND STALL PATTERNS

R.N. = 3.7×10^6
FLAPS 55°
AIRFLOW NACELLES



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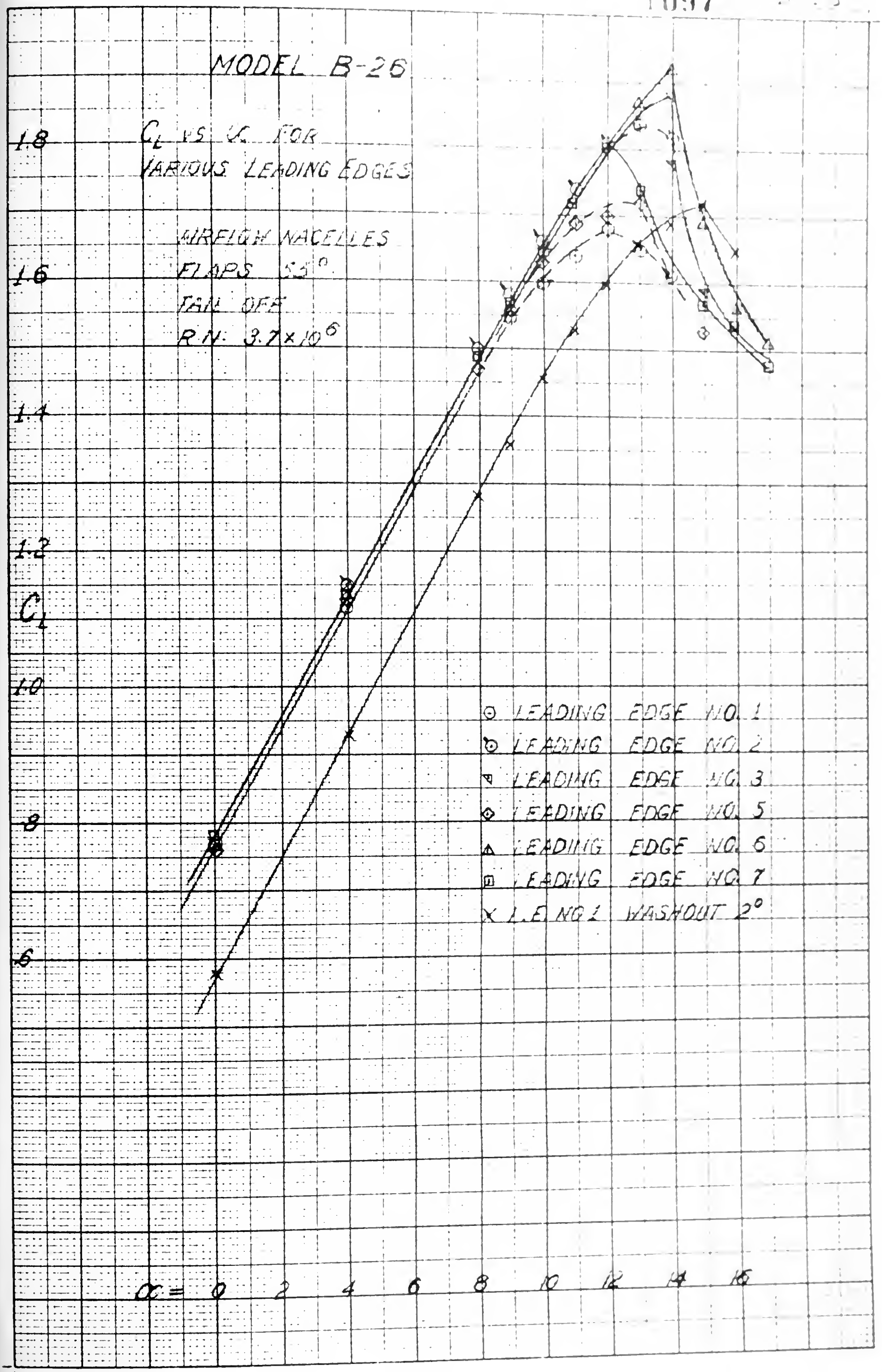
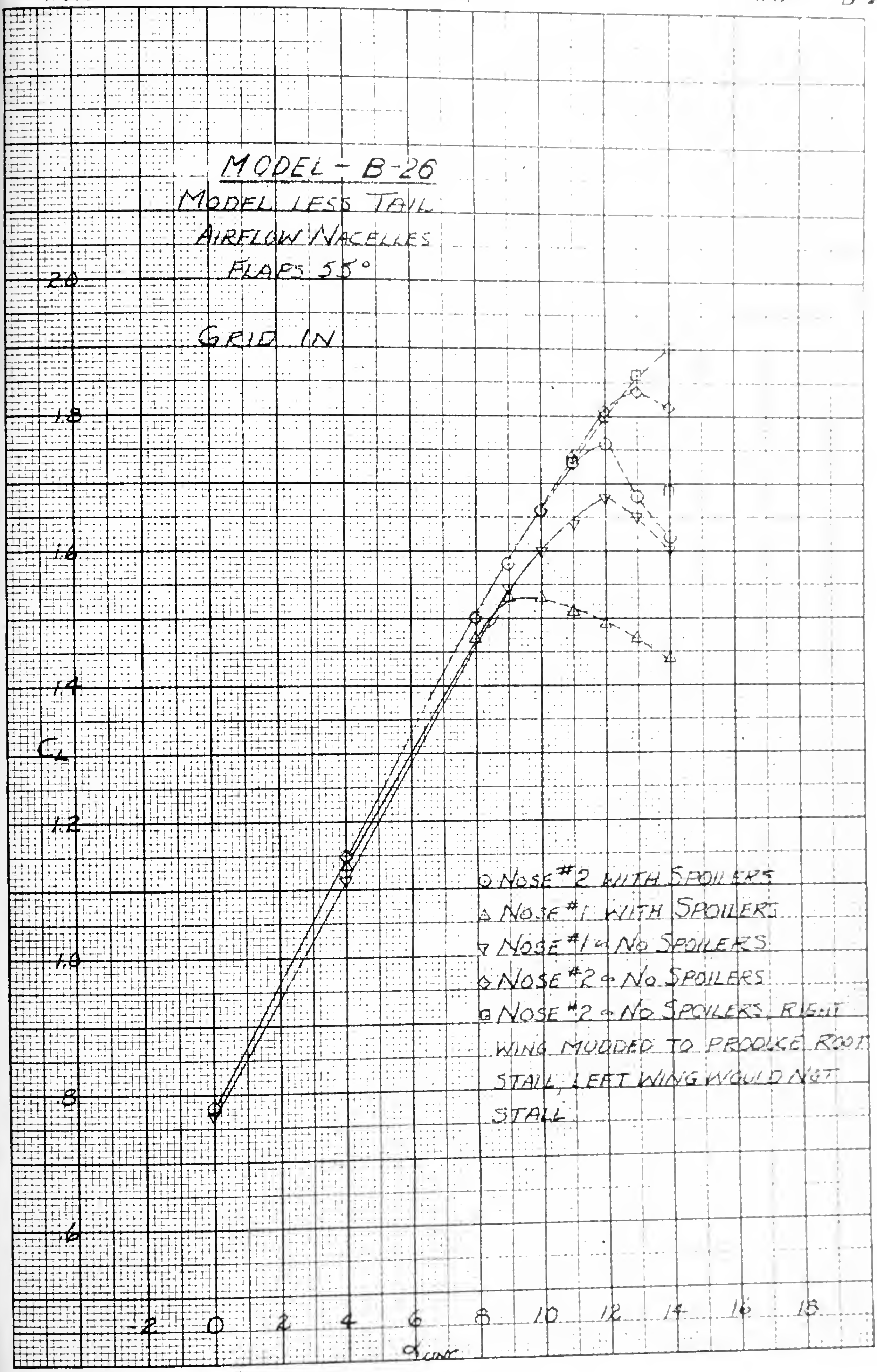
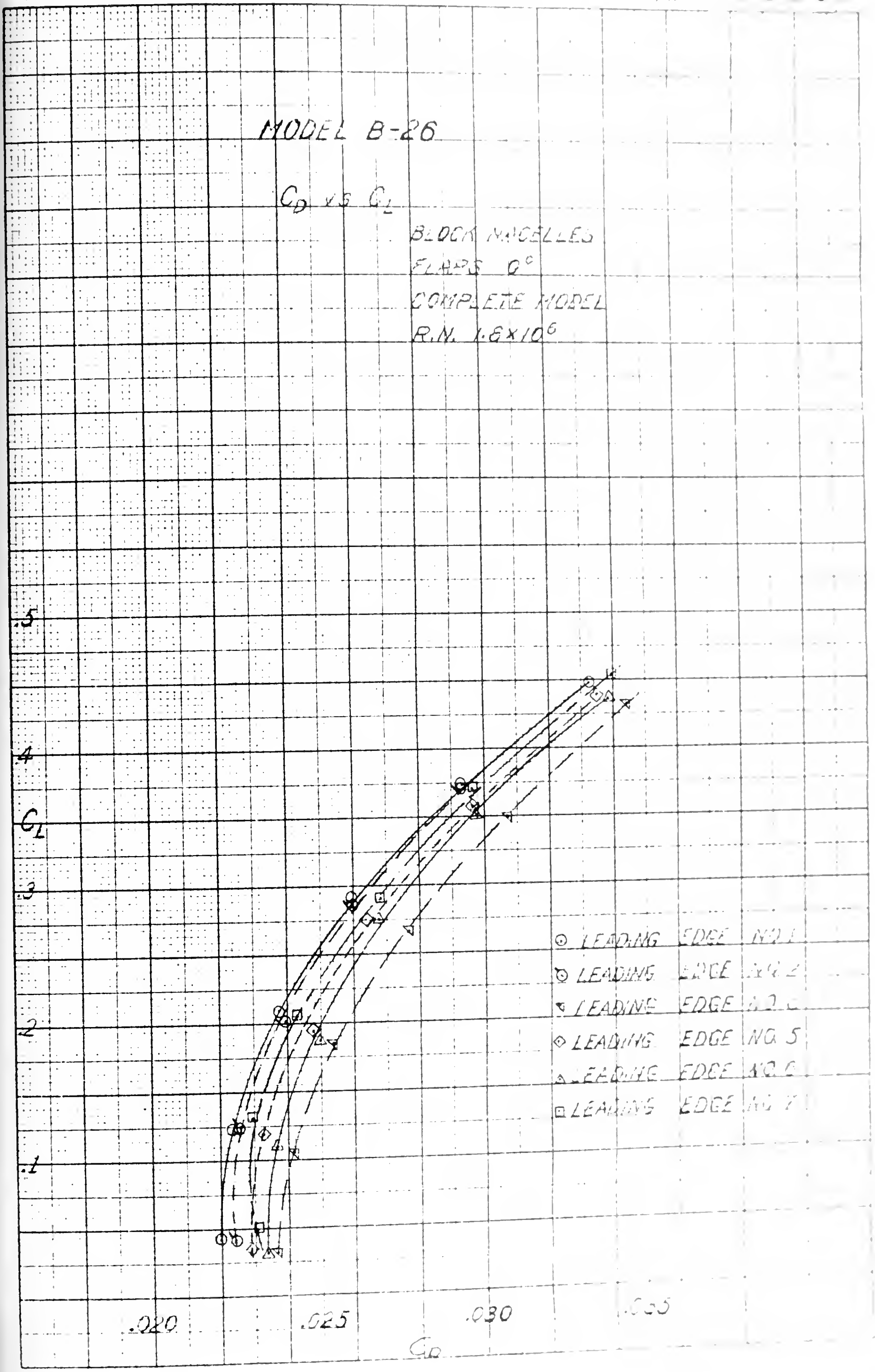


MODEL B-26
C_L VS α FOR
VARIOUS LEADING EDGES
AIRFLOW NACELLES
FLAPS 55°
TAIL OFF
R.N. 3.7×10^6
1.8
1.6
1.4
1.2
C_L
1.0
.8
.6
LEADING EDGE NO. 1
LEADING EDGE NO. 2
LEADING EDGE NO. 3
LEADING EDGE NO. 5
LEADING EDGE NO. 6
LEADING EDGE NO. 7
L.E. NO 1 WASHOUT 2°
α = 0 2 4 6 8 10 12 14 16


7/19-40
MODEL - B-26
MODEL LESS TAIL
AIRFLOW NACELLES
FLAPS 55°
GRID IN
2.0
1.8
1.6
1.4
1.2
1.0
.8
.6
C_L
-2
0
2
4
6
8
10
12
14
16
18
α_{unc}
NOSE #2 WITH SPOILERS
NOSE #1 WITH SPOILERS
NOSE #1 & NO SPOILERS
NOSE #2 & NO SPOILERS
NOSE #2 & NO SPOILERS, RIGHT WING MUDDED TO PRODUCE ROOT STALL, LEFT WING WOULD NOT STALL


MODEL B-26
C_D vs C_L
BLOCK NACELLES
FLAPS 0°
COMPLETE MODEL
R.N. 1.6×10^6
LEADING EDGE NO. 1
LEADING EDGE NO. 2
LEADING EDGE NO. 3
LEADING EDGE NO. 5
LEADING EDGE NO. 6
LEADING EDGE NO. 7
C_L
.1
.2
.3
.4
.5
.020
.025
.030
.035
C_D

Prepared By D.S. Lewis Jr.
Date 7-9-40

THE GLENN L. MARTIN COMPANY
BALTIMORE, MD.

1100

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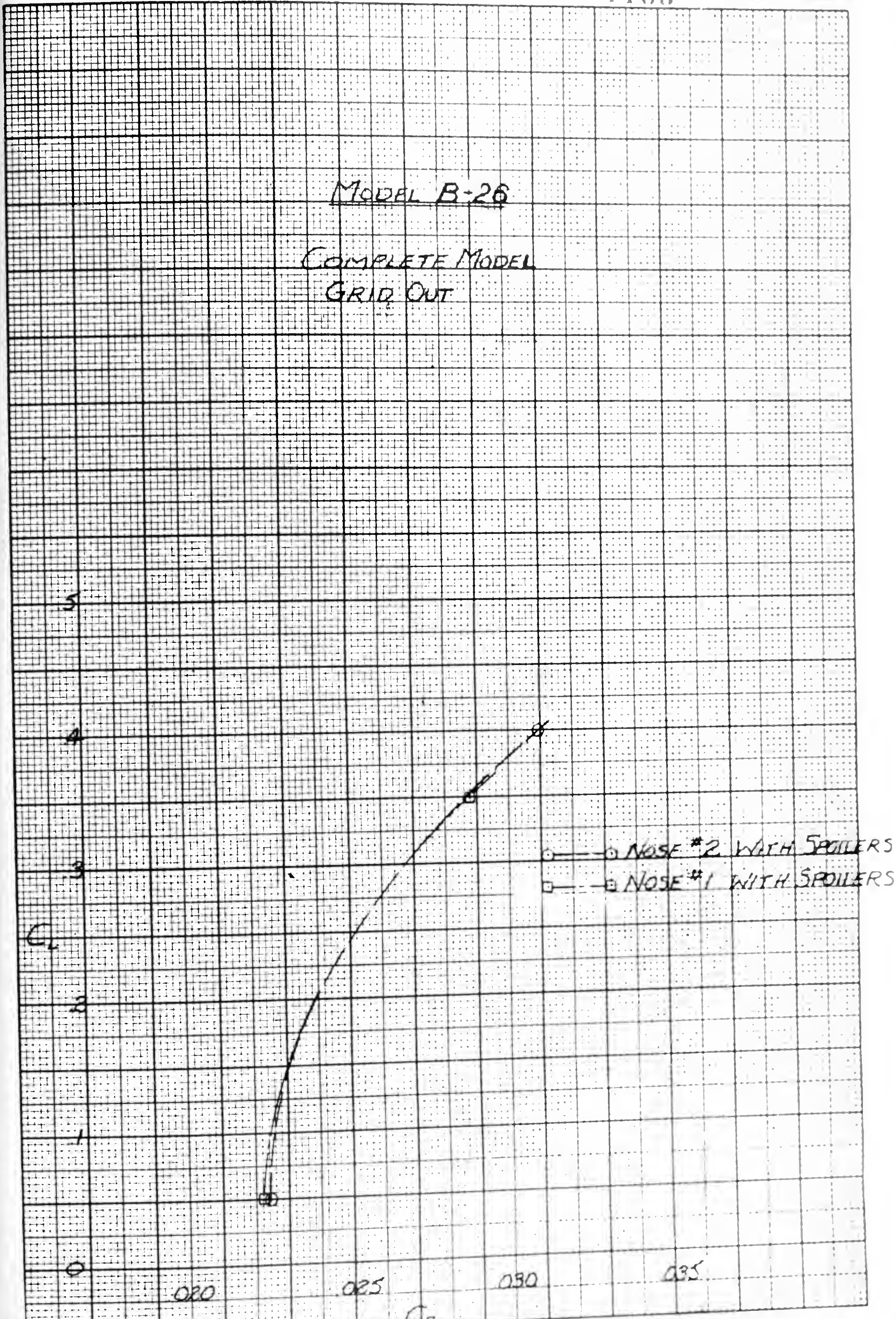
Model B-26



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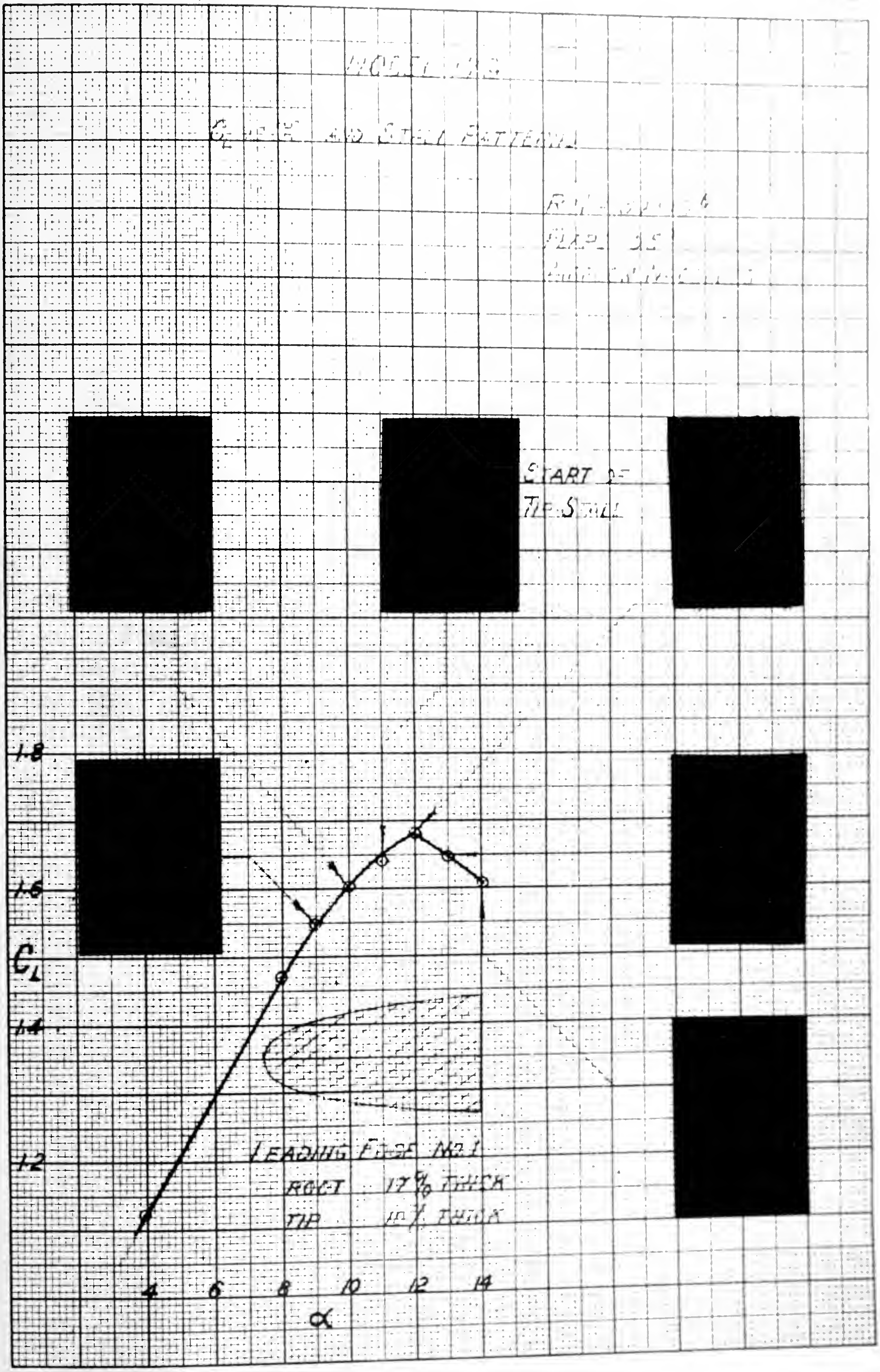
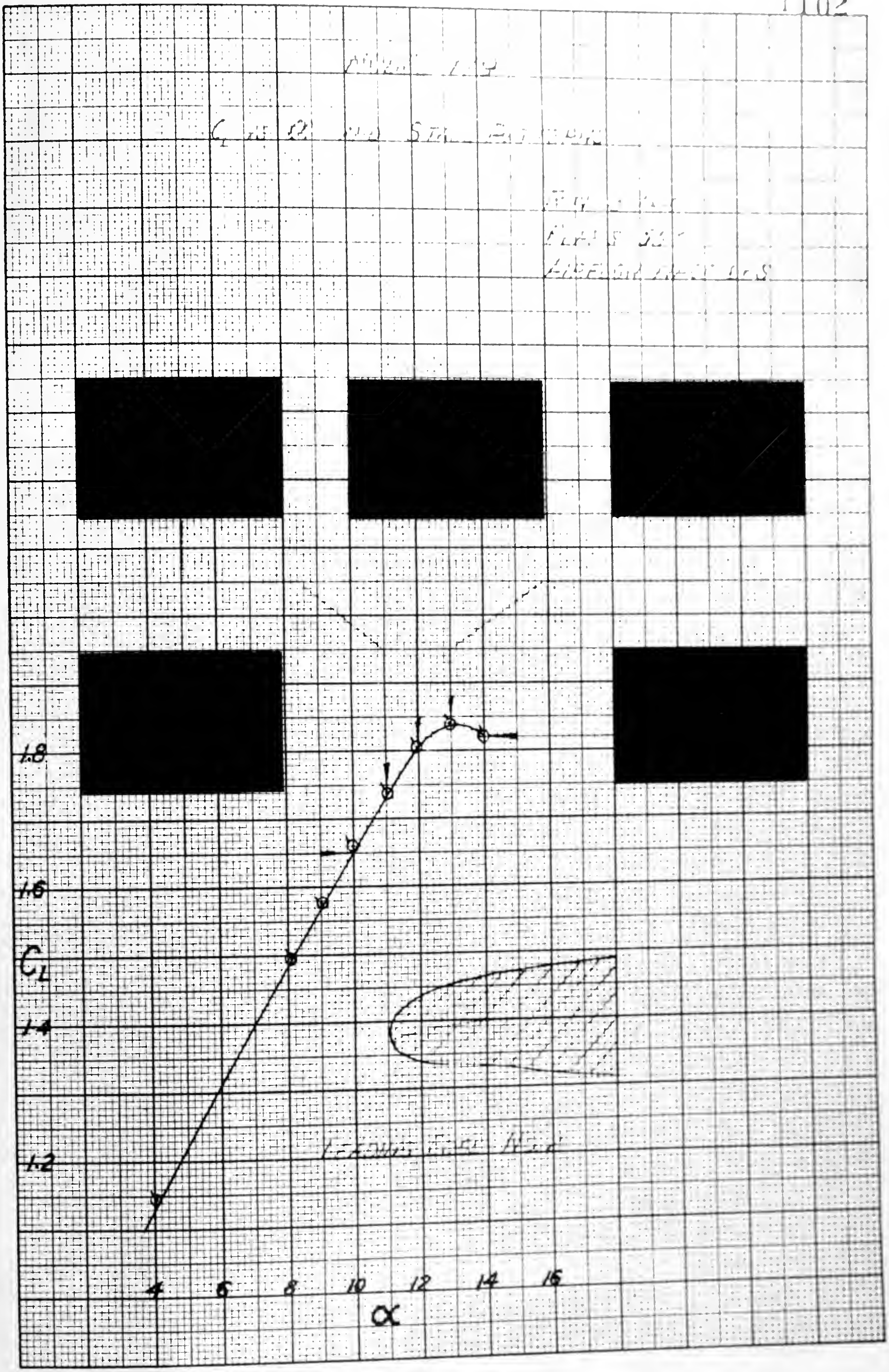
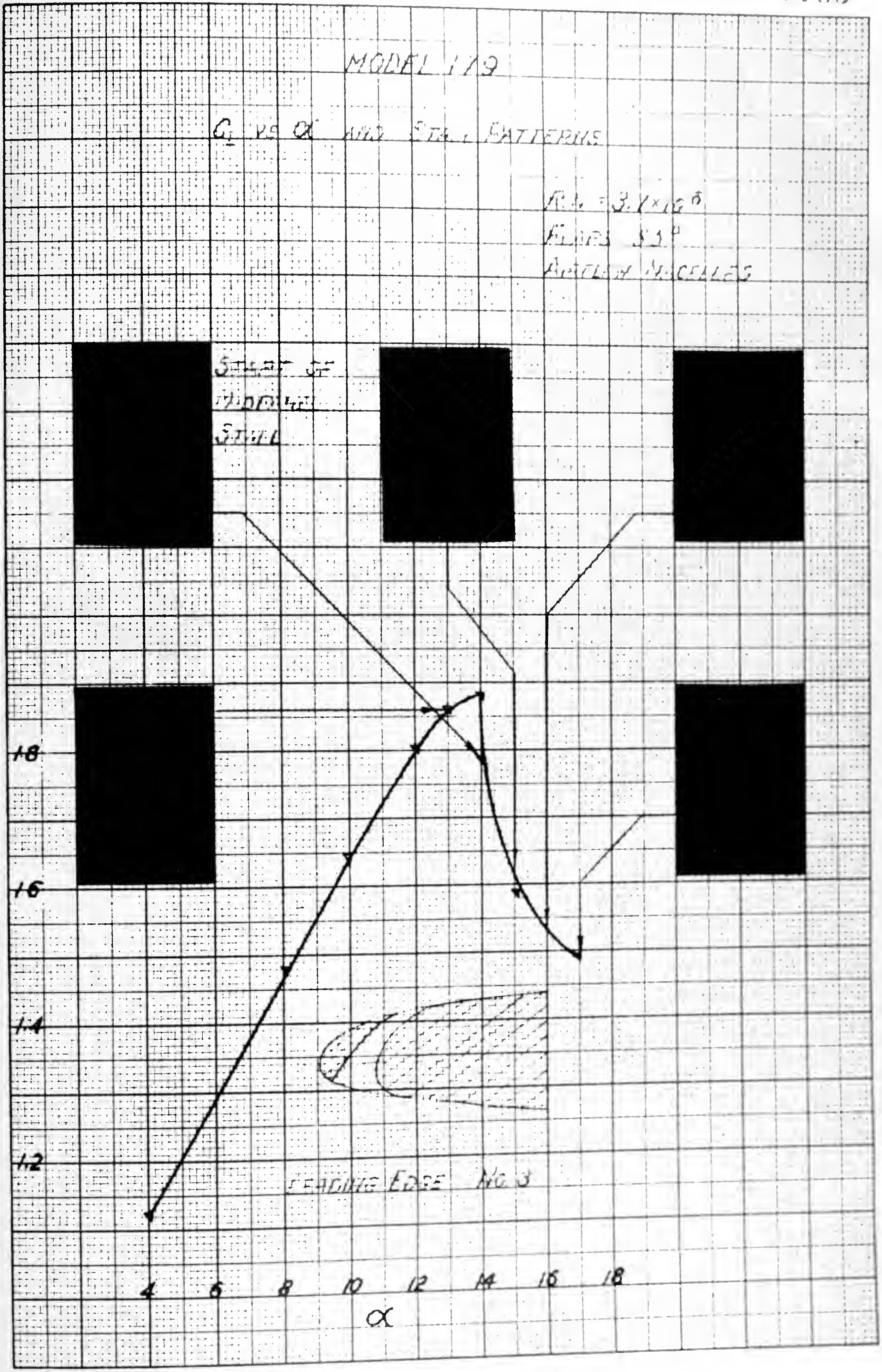

START OF
TIP STALL
1.8
1.6
C_L
1.4
1.2
LEADING EDGE No.1
ROOT 17% THICK
TIP
4
6
8
10
12
14
α


1.8
1.6
C_L
1.4
1.2
4
6
8
10
12
14
16
α


MODEL 179
1.8
1.6
1.4
1.2
LEADING EDGE No. 3
4
6
8
10
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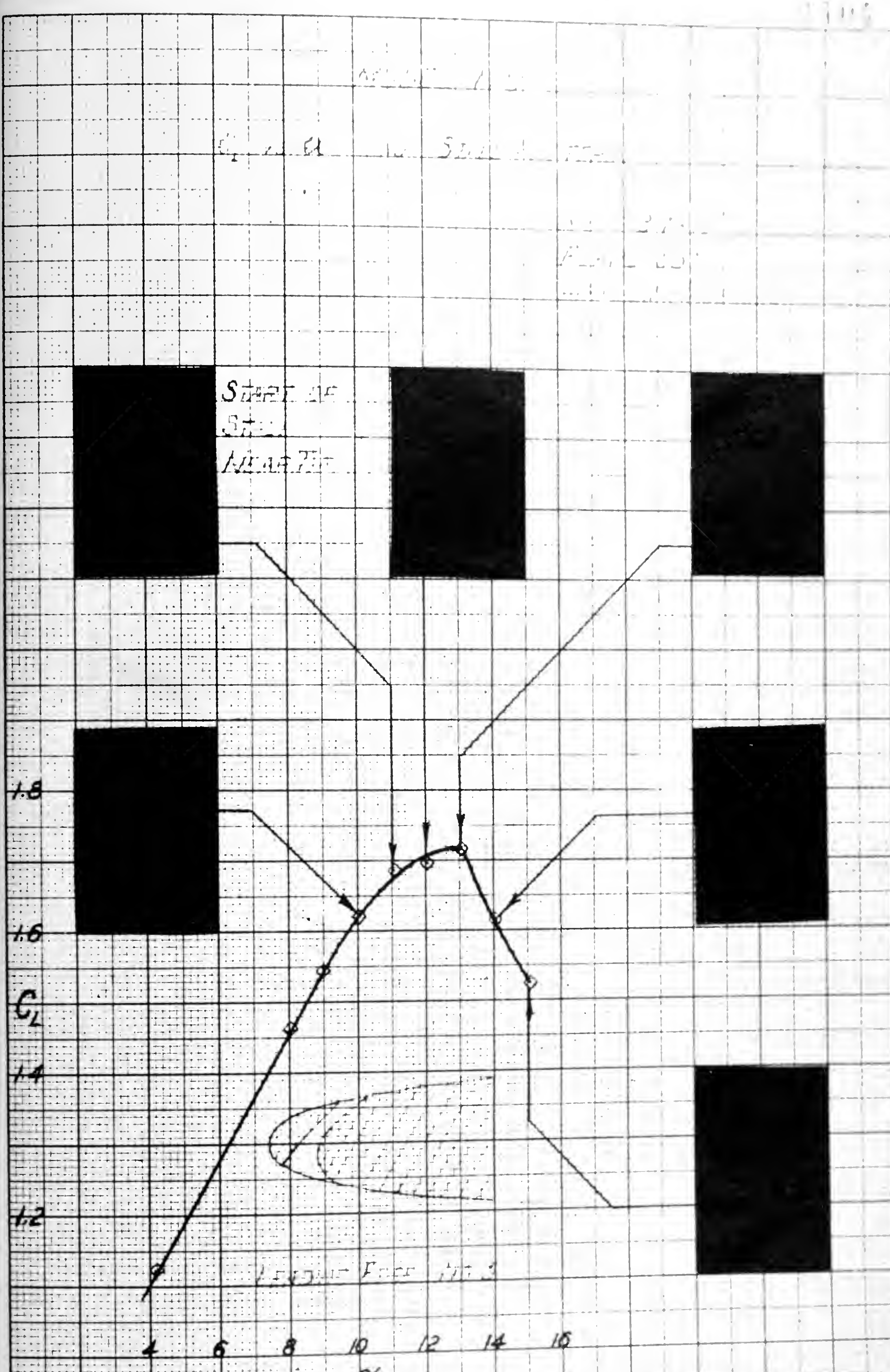

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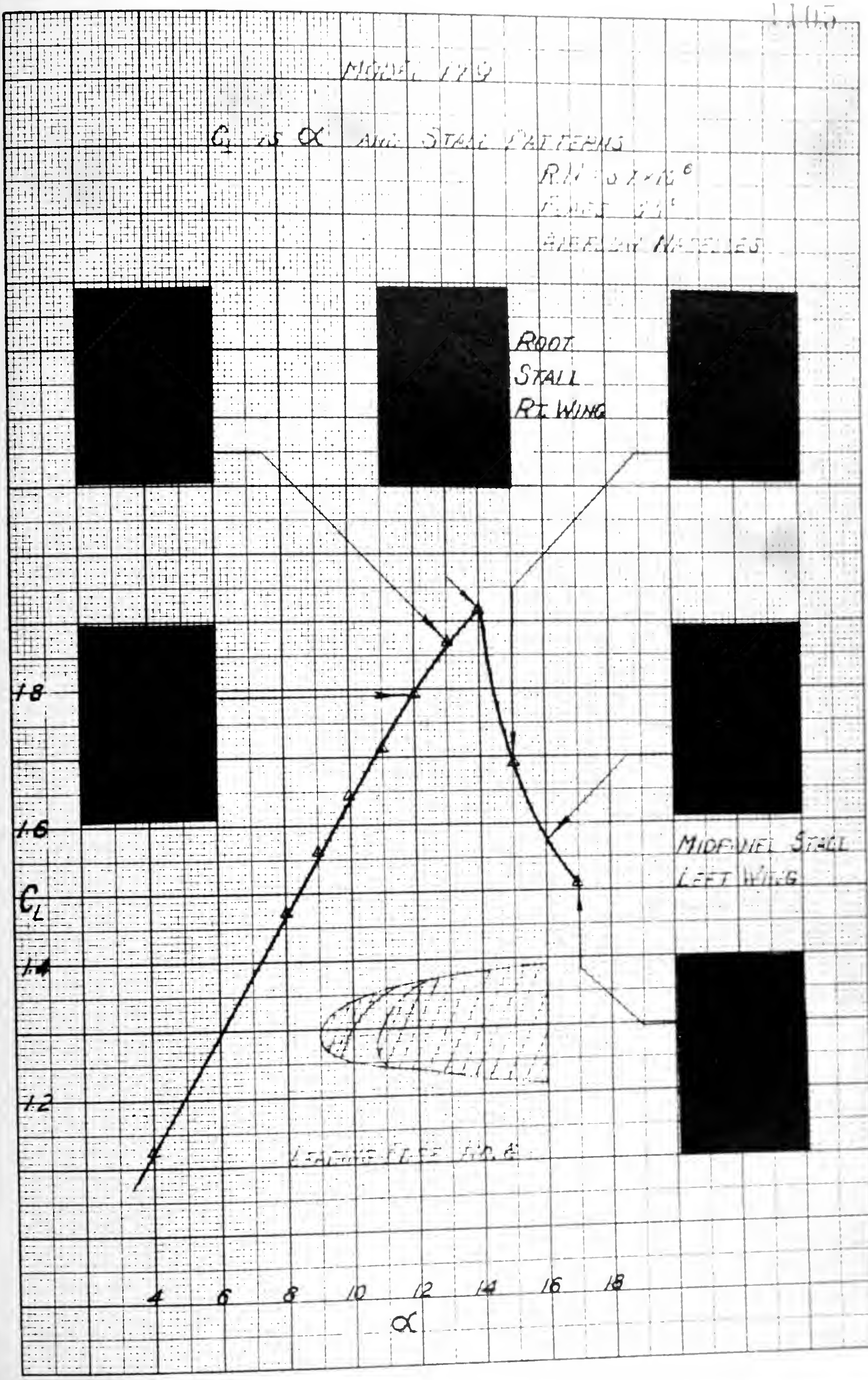

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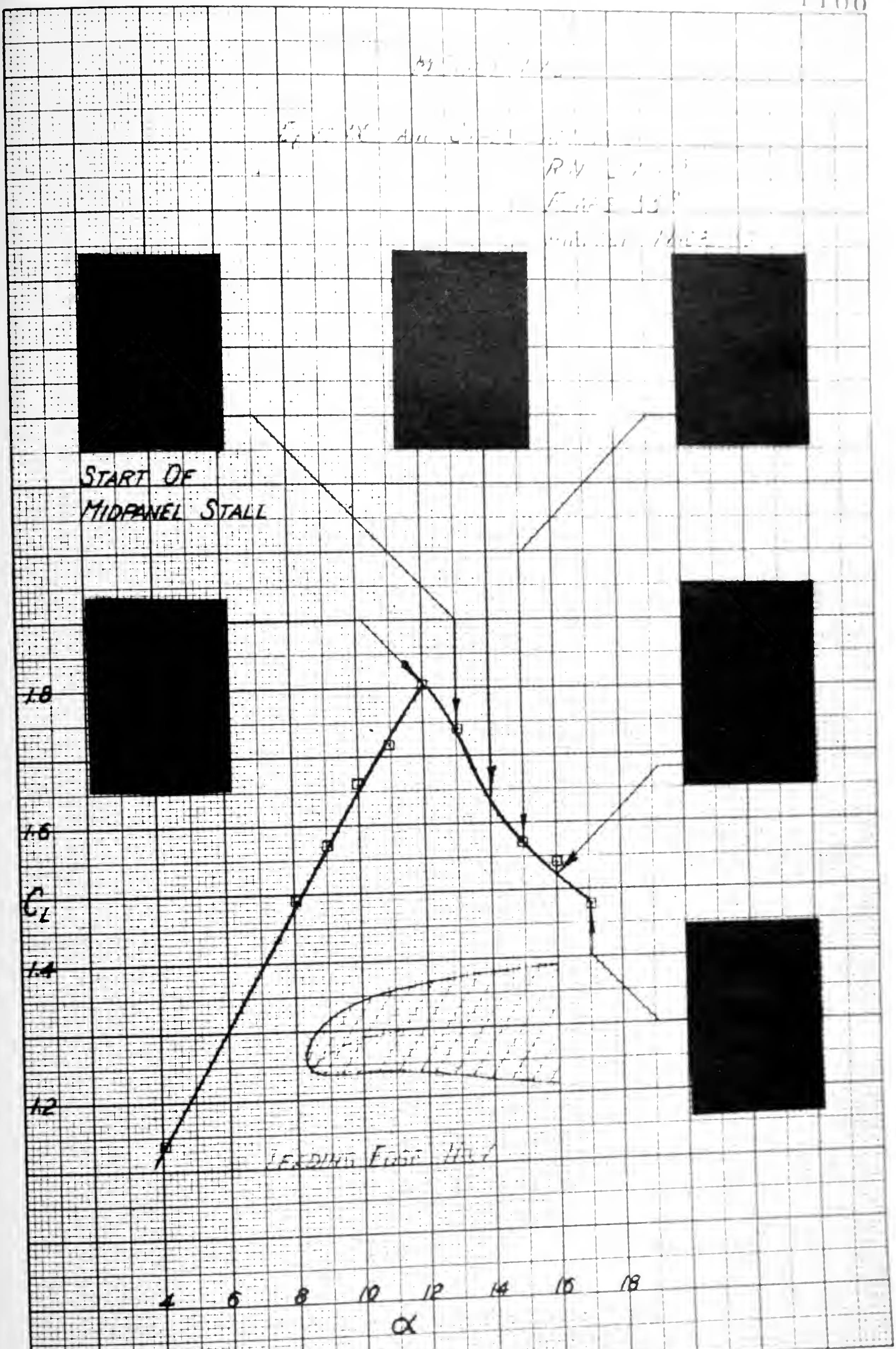

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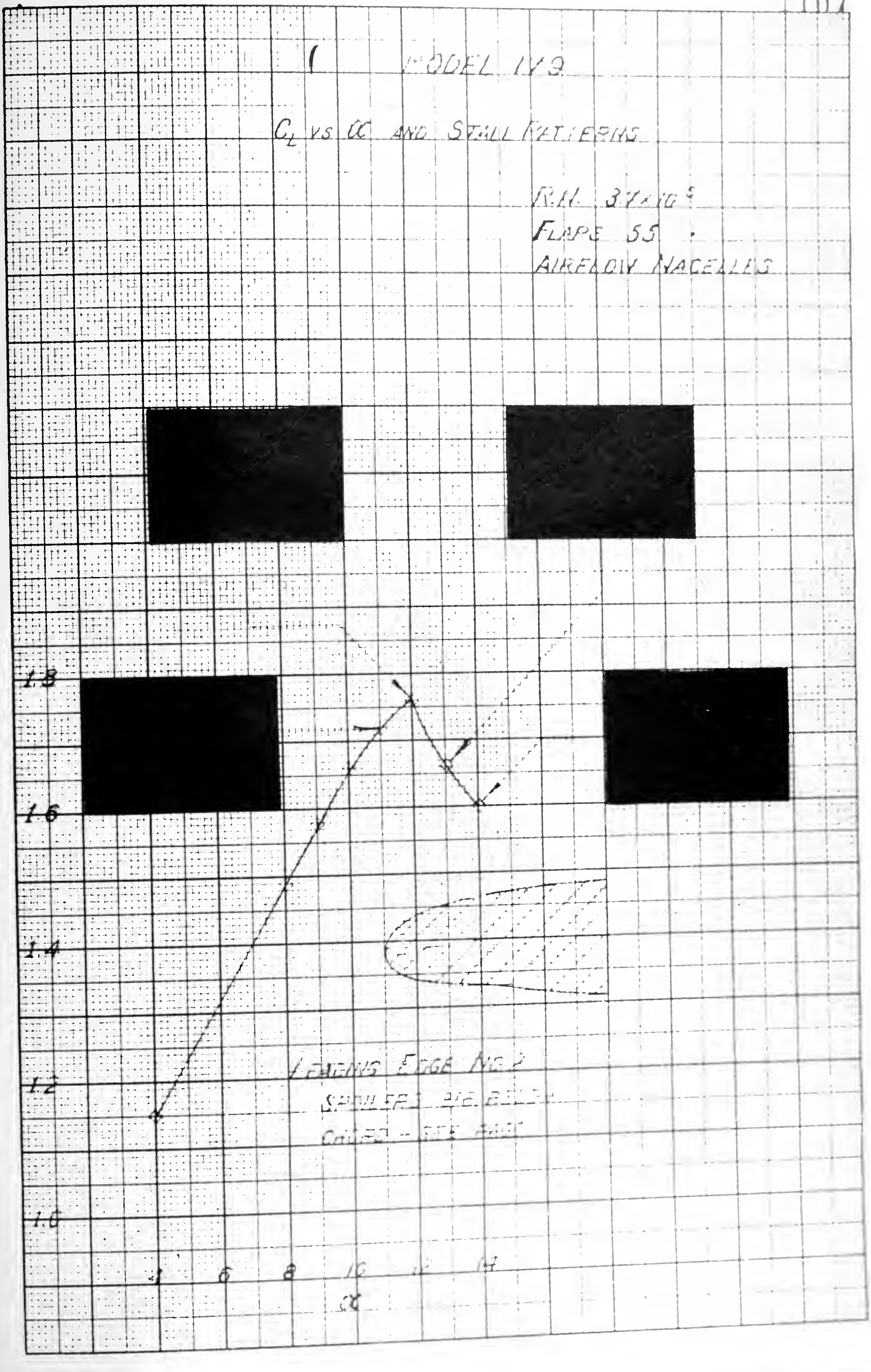
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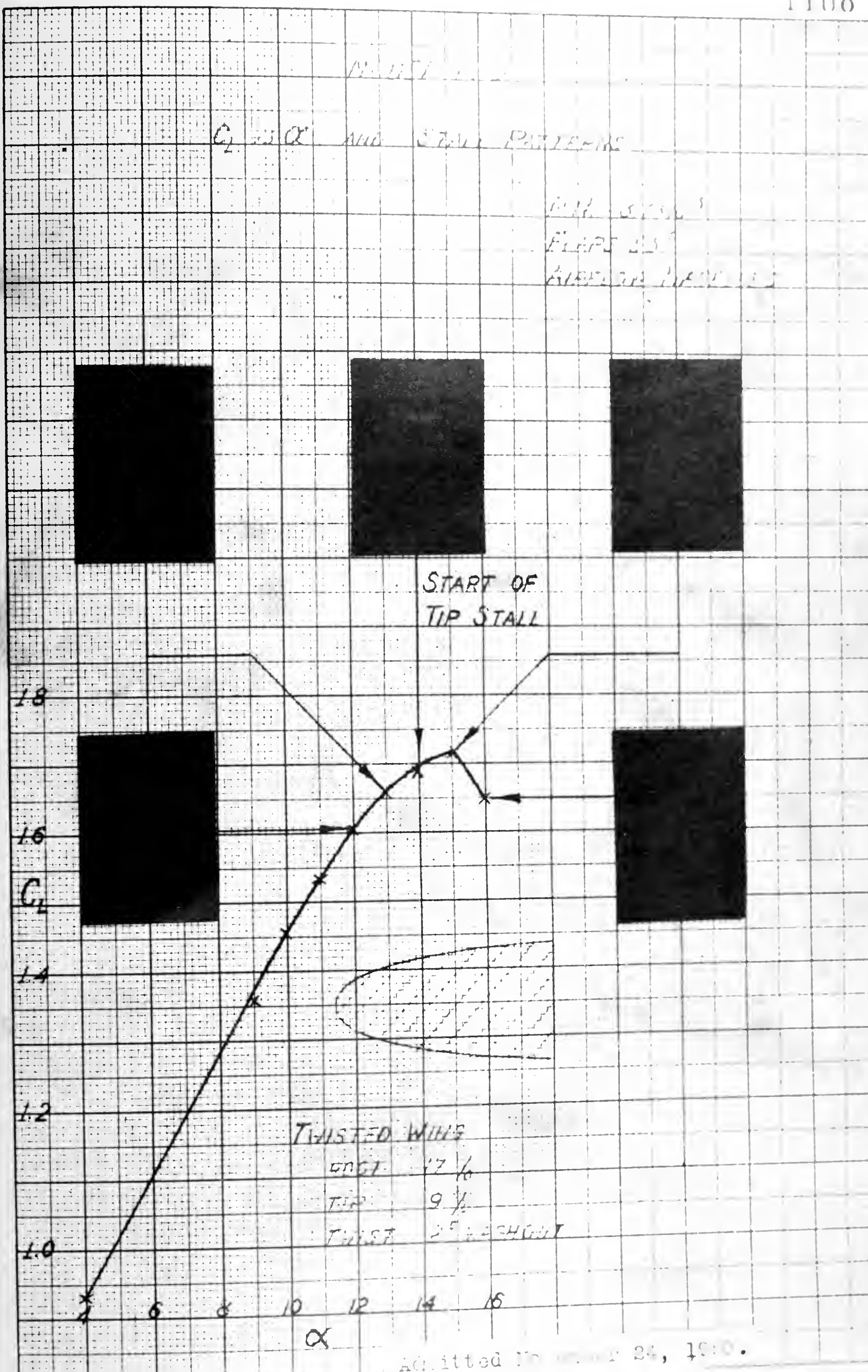
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In the United States Court of Appeals
for the Ninth Circuit

No. 12885

CONSOLIDATED VULTEE AIRCRAFT CORPORATION, a Delaware Corporation, and AMERICAN AIR LINES, INC., a Delaware Corporation,

Appellants,

vs.

MAURICE A. GARBELL, INC., a California Corporation, and GARBELL RESEARCH FOUNDATION, a California Corporation,

Appellees.

STIPULATION AND ORDER

It Is Hereby Stipulated by and between appellants and appellees under the provisions of Rule 76 (h) that the record on appeal may be supplemented to include the following material omitted from the record on appeal:

(a) The affidavit of Theodore Roche, Jr., executed January 31, 1951, filed in opposition to Defendants' Motion for New Trial.

(b) The affidavit of Maurice A. Garbell, executed January 30, 1951, filed in opposition to Defendants' Motion for New Trial.

(c) This stipulation.

And that this stipulation constitute a designation of the supplemental record to be printed as a supplement to the record heretofore filed in this cause and that the attached constitute true copies of the affidavits of Theodore Roche, Jr., and Maurice A.

Garbell hereinabove identified and that the supplemental record so designated by this stipulation may be printed and will constitute a supplement to the record on appeal.

This stipulation is entered into at the request of appellees, and appellants consent thereto only upon the condition that their time for filing their opening brief on appeal be continued and reset to commence upon the clerk's mailing to appellants copies of the printed supplement to the printed record referred to in this stipulation, such time to expire not earlier than September 16, 1951.

It Is Further Stipulated that the cost of printing the supplement referred to herein shall be borne by appellees.

Dated August 14, 1951.

HARRIS, KIECH, FOSTER & HARRIS,
/s/ FORD HARRIS, JR.,
Attorneys for Appellants.

LYON & LYON,
/s/ LEWIS E. LYON,
Attorneys for Appellees.

So Ordered:

/s/ WILLIAM DENMAN,
Judge of the United States Court of Appeals for the Ninth Circuit.

/s/ CLIFTON MATHEWS,
/s/ H. T. BONE,
Judges U. S. Court of Appeals for the Ninth Circuit.

District Court of the United States, Southern District of California, Central Division

Civil Action No. 10930-Y

MAURICE A. GARBELL, INC., a California Corporation, and GARBELL RESEARCH FOUNDATION, a California Corporation,

Plaintiffs,

vs.

CONSOLIDATED VULTEE AIRCRAFT CORPORATION, a Delaware Corporation, and AMERICAN AIR LINES, INC., a Delaware Corporation,

Defendants.

AFFIDAVIT OF THEODORE ROCHE, JR.

State of California,
County of Los Angeles—ss.

Theodore Roche, Jr., being first duly sworn, deposes and says:

That at all of the times herein mentioned affiant was, and is now, one of the attorneys of record for the plaintiffs in the above-entitled action, and as such has read and is familiar with defendants' Motion for a New Trial, together with supporting affidavits hereinbefore filed herein.

Addressing himself to the grounds of said Motion for New Trial of (1) surprise and (3) newly discovered evidence, affiant states the following:

Trial of this action commenced at 10:00 a.m., Tuesday, November 21, 1950. Prior thereto and on the 3rd day of August, 1950, by stipulation, defendants took the deposition of Maurice A. Garbell, the inventor of the patent involved herein. Said witness was questioned by Mr. Fred Gerlach, one of defendants' attorneys of record, who at all times during the taking of said deposition was assisted by Mr. Glendon T. Gerlach, the patent director of Consolidated Vultee Aircraft Corporation.

No restrictions or limitations of any kind were placed upon the examination of said Maurice A. Garbell, and said defendants, through their counsel, were afforded full opportunity to, and had they so desired, could have questioned the said Maurice A. Garbell fully, completely and in detail concerning all of the matters, and each of them, ultimately testified to by him during the trial of said action, and by such questioning could have ascertained the name and whereabouts of each person referred to by Mr. Garbell in said testimony, including those individuals named in defendants' Motion for New Trial, to wit: Harry B. Chin, Theodore P. Hall and Donald A. Hall.

This action was commenced in January, 1950. Long prior thereto defendants fully knew that the said Theodore P. Hall and Donald A. Hall were employed by defendant, Consolidated Vultee Aircraft Corporation during the entire period of employment of Maurice A. Garbell by said last named defendant, and that the said Theodore P. Hall and Donald A. Hall were possessed of knowledge which had direct bearing upon the activities of the said

Maurice A. Garbell during the period of his employment by defendant, Consolidated Vultee Aircraft Corporation, with relation to the subject matter of the invention referred to herein.

In the year 1948 affiant was engaged in investigating the truth or falsity of the facts as related to him by Maurice A. Garbell in order to determine whether or not to accept employment in a proposed action against defendants herein based upon the alleged infringement of the patent in suit. In the course of such investigation affiant had several conferences with Mr. Glendon T. Gerlach, who then was and is still Patent Director for Defendant, Consolidated Vultee Aircraft Corporation. Among other things affiant informed Mr. Gerlach that Mr. Garbell claimed that he had suggested the use of his patented wing to defendant, Consolidated Vultee Aircraft Corporation, at every opportunity during a period commencing within a few weeks after the start of his employment by said defendant and lasting during the entire term of said employment.

On or about the 21st day of July, 1948, at the prior suggestion of the said Mr. Glendon T. Gerlach, affiant visited the plant of defendant, Consolidated Vultee Aircraft Corporation, at San Diego, California, the said Mr. Gerlach and affiant together then interviewed and questioned the following persons: Theodore P. Hall, Donald A. Hall, Ralph Bayless and Kenneth E. Ward, all of whom were then and there working at the said plant of defendant, Consolidated Vultee Aircraft Corporation. Each individual was interrogated by affiant and by Mr. Gerlach as to his knowledge of the patented Garbell wing and the suggestions of its use as made by Mr. Garbell during the term of his employment by defendant, Consolidated Vultee Aircraft Corporation.

Upon the conclusion of said interviews, the said Glendon T. Gerlach made a statement to affiant in substantially these words:

"At the outset I was sure Garbell had never mentioned his wing, but after hearing the men today I am convinced Garbell tried to push the use of his wing at every opportunity."

In the early part of August, 1948, the said Mr. Gerlach and affiant had a further interview with Donald A. Hall at the plant of said defendant, Consolidated Vultee Aircraft Corporation, relating to the same subject matter. During one of the conferences held between affiant and the said Glendon T. Gerlach there was placed in affiant's hand by Mr. Gerlach a copy of an analysis of the Garbell patent, which analysis was signed by the said Donald A. Hall and which has been introduced in evidence in this case by plaintiffs as their Exhibit 21.

The Ralph L. Bayless and Kenneth E. Ward above referred to testified on behalf of defendants at the trial of this action. The said Glendon T. Gerlach, Patent Director of defendant, Consolidated Vultee Aircraft Corporation, assisted defendants' counsel in the preparation and trial of this action and he had actual knowledge of the connection between Mr. Garbell, the inventor, Theodore P. Hall, Donald A. Hall and defendant, Consolidated Vultee Aircraft Corporation, as hereinabove set forth. Defendants did not call the said Theodore P. Hall and Donald A. Hall as witnesses.

On the 3rd day of July, 1948, while investigating the facts of this case as hereinabove set forth, for the first time affiant interviewed Mr. Harry B. Chin, and upon that occasion took a statement from him. It had been explained to Mr. Chin that a statement from him was desired upon the ground that Mr. Garbell was dealing with a potential licensee of his patented wing and that we desired to ascertain if there was proof of invention prior to the employment of Mr. Garbell by the potential licensee. Said statement was taken by affiant at his office, not in the presence of Mr. Garbell, was voluntarily given by Mr. Chin; said statement was taken down in shorthand by the secretary of affiant, thereafter transcribed in the office of affiant, such transcription being as follows:

"Mr. Roche: What is your full name?

"Mr. Chin: Harry Bradford Chin.

"Q. And the address?

"A. My present address is 715 Commercial Street—that is where I pick up all my mail—my family live there; although I have an apartment of my own at 1060 Powell Street.

"Q. At the present time you are employed?

"A. By United Airlines.

"Q. In the San Francisco office?

"A. Yes, at Mills Field.

"Q. In what capacity?

"A. Aerodynamic performance engineer.

"Q. You know Dr. Garbell? A. Yes.

"Q. Do you recall when you first met him?

"A. I first met him when I was working in the Boeing School already, and Dr. Garbell came right after a Mr. Thorpe left, which I guess was in November, or thereabouts—October or November—of 1939, or thereabouts, I believe.

"Q. The Boeing School that you refer to is located here?

"A. Yes, at Oakland Municipal Airport.

"Q. Tell me, was that school established by Boeing Aircraft Corporation, or was it established by the Government?

"A. It was established by—through donations of W. E. Boeing, way back in 1929, before the consolidation of airlines, which was later called United Airlines—part of Boeing Transport and Boeing Air Company—before the mail cancellation in 1934.

"Q. It was established as sort of a foundation, by Boeing personally, from his own funds, I gather?

"A. Yes.

"Q. And the purpose of the school was what? They instructed and——

"A. Yes. You might say it is a trade school, and it is a source from which Boeing Air Transport and National Air Transport and quite a few of the airlines draw their personnel—their mechanical personnel.

"Q. You went there first when?

"A. I went there first as a student in 1934.

"Q. To become an engineer?

"A. At that time there was no engineering course. I took what they call a Master Mechanic course and Design Subjects, and so forth. It is a regular mechanics school as well as a flying school.

"Q. They taught flying also? A. Yes.

"Q. That was in 1934? A. Yes.

"Q. At that time were the air lines drawing on that school for their personnel?

"A. Yes, considerably, because there is a placement bureau opened by Boeing School, which helped the graduate to obtain jobs in the industry.

"Q. Did they charge a student going there? Did he have to pay for his tuition? A. Yes.

"Q. It was a regular trade school, in the accepted sense? It wasn't maintained by the air lines? They didn't pay——

"A. No, it was self-supporting.

"Q. You started there in 1934, and as a student you were at the school how long?

"A. One year. Not quite one year. In fact, the course was a nine-month course and I graduated and then I took a couple of months of postgraduate work, so making it, all in all, eleven months. Then I became an instructor in aeronautics at the same school.

"Q. You became an instructor the latter part of 1934, or 1935?

"A. No. The latter part of 1935.

"Q. And you were instructing in what capacity?

"A. At first—the first few months I was a reader in the Aerodynamics and Strength of Material Department, as well as assistant instructor in Drafting and Designing.

"Q. That latter subject—was that the drafting and designing—was that airplanes or planes as a whole, or wing design or body design?

"A. Generally complete ships.

"Q. Did you continue in that particular field, or did you progress into other subjects, and between 1935 and 1939—there is a four-year period—you remained at the school?

"A. Yes. During those four years, while I taught a variety of subjects, including mathematics and aerodynamics and mechanical design and illustrative and descriptive geometry.

"Q. Along the latter part of 1939 you say there was a Mr. Thorpe. Was he an instructor?

"A. Yes.

"Q. And he left there? A. Yes.

"Q. And Dr. Garbell came to take his place?

"A. Not exactly to take his place; you might say as far as the lecture material was concerned. While the instructors do teach the various material, and when Mr. Thorpe left there was an opening, obviously, and I believe Mr. Garbell was hired on that open requisition.

"Q. There was an opening and he was employed, as far as you know, to fill it?

"A. Yes. Because I took over most of Thorpe's subjects after Thorpe left, which was mainly design.

"Q. Plane design? A. Yes.

"Q. I suppose it is true, Mr. Chin, that during those four years, in connection with the field of plane design, that you had given a lot of attention and a considerable part of your work dealt with wing design, and the structure of wings, and the air forces? That is true, isn't it?

"A. That is right.

"Q. Did you teach or lecture students on those subjects?

"A. Yes. Simultaneously during those four years Mr. Thorpe and myself designed two airplanes for Boeing School of Design, and both those airplanes were built by the school and flown by the school.

"Q. What type, single motor?

"A. Single-motor, two-seater trainers.

"Q. Did you embody any new principles of design in those planes, either in the wing, or in any fashion, from what had preceded the trainers? There was some change, wasn't there?

"A. There are changes going on at all times, due to past knowledge. I would say the airplanes that we designed were strictly conventional types, because we designed it as a trainer; so, therefore, any characteristics of the airplane should be, of necessity, conventional, and those characteristics are known; so that the airplane, when done, would be an honest, conventional, othodox airplane?

"Q. There wouldn't be any radical change in it, then. Is that correct? A. Yes.

"Q. But in the field of aerodynamics and the designing of planes, and wings in particular—during those four years you gave great study to different types of wing construction? A. Yes.

"Q. And principally dealing with the effect of air flow over the wing surface, isn't that correct?

"A. Yes.

"Q. Now at the time there already had been some well-known patents issued on or covering wing design? A. Yes, there were.

"Q. For many different types of wings; but there were some major types which were in general use. Isn't that correct? A. Yes.

"Q. And you were familiar with them?

"A. Yes.

"Q. Now, tell me, Mr. Chin, it is true, isn't it, that the goal toward which a plane designer goes is to design a wing which has very good stalling characteristics? A. Yes.

"Q. In other words, everybody designing a plane, or a wing for a plane, for general use attempts to eliminate, if they can, stalling characteristics. Is that right?

"A. No. I do not believe that is a correct statement, because I do not believe you can entirely eliminate stalling. You might say we try to eliminate any violent characteristics accompanying a stall, and, if possible, have sufficient warning before a stall.

"Q. In other words—let me put it this way: the result which you would seek to achieve in designing the wing would be that in performance violent results would tend to be eliminated from the stalling characteristics, first—— A. Yes.

"Q. And, secondly, or as a part of it, the design in operation would cause the wing to give a warning that a stall was approaching. Is that correct?

"A. That's right. You might put it that way, more specifically: an airplane that has honest stalling characteristics should be designed such that a stall is unaccompanied by any rolling motion of the airplane, and that can be done—whatever the means is a different story—by moving the point at which the complete wing first stalls—by moving this point inboard, closer to the fuselage. If the initial stalling point is out toward the wing tip, obviously any stall would be accompanied by a rolling motion of the airplane, and if the stall point is inboard or closer to the fuselage of the airplane, then when the airplane does stall, it will stall and fall straight ahead, unaccompanied by an violent rolling motion. In other words, it will just pitch until its usefulness is again obtained, by pitching of the nose downwards.

"Q. In other words, the nose would pitch downwards, so that the plane would tend to drop, and thereby gain speed, so that the stall of the wing would be again overcome. A. That's right.

"Q. In the case where the stall is accompanied by a very violent rolling motion and a plane does get into a stall, does a spin result in the plane?

"A. Generally, yes.

"Q. And then that is almost impossible to pull out of, is that right, in these larger planes?

"A. Not exactly; if the airplane is what we call dynamically stable, the airplane will come out of a spin, with the controls neutral, by itself within 1½ rolls. You might say if the airplane has gotten into a spin and the controls are neutralized, and the hands and feet are off the controls, the airplane should be able to pull out of a spin within 1½ rolling motions of the airplane—and by itself; if the airplane were dynamically stable. Of course, you do have the catastrophic type, that gets worse and worse as it spins, but if the airplane were designed correctly it would come out of it.

"Q. Had you known Dr. Garbell, or known or heard of him, prior to his coming to the Boeing School? A. No, I had not.

"Q. So that the first time you ever heard of him or met him was after he became employed as an instructor at the school? A. Yes.

"Q. This was in 1939? A. Yes.

"Q. At that time, with the general world conditions being what they were, had the government stepped into the picture in any way in connection with that school? A. No.

"Q. However, due to certain security rules and regulations which were in existence, a person who was not a citizen of the United States could not work or have any connection with any of the airplane manufacturers and buiders at that time. Is that true? Were you aware of that?

"A. I believe that has been the practice of all the major companies, to hire only citizens or persons who have taken out first papers.

"Q. Let's say this: Dr. Garbell was at the school from October or November of 1939, according to your recollection, until when? About?

"A. He left to join Pan-American Air Ferry Group, let us say—I would say somewhere in 1941.

"Q. To the best of your recollection, in round numbers, he was at the school approximately two years, we will say. That is correct? A. Yes.

"Q. During that two-year period, did you become acquainted with the doctor? A. Yes.

"Q. Were you working together in connection with any projects of the school?

"A. Not any particular project, no; but as far as teaching courses, yes. In other words, for instance, when Dr. Garbell left—he left in the middle of the semester, as it were—I took over a couple of his courses.

"Q. What courses did you take over?

"A. I took over the differential calculus course from him and also the strength of materials course from him.

"Q. Did Dr. Garbell teach or lecture in connection with a course on plane or wing design, do you recall?

"A. I was the chief instructor in design at the time he left, although Dr. Garbell taught some aerodynamic courses—which ones I don't recall.

"Q. Now, tell me, Mr. Chin, during the time that Garbell was at the school there, did you become pretty well acquainted with him? A. Yes.

"Q. And I suppose that in connection with your work you had frequent discussions of problems, is that right? A. Yes.

"Q. And, of course, you both were interested in everything connected with aerodynamics and planes, that's true? A. Yes.

"Q. And during that time I suppose you had many, frequent discussions and conversations concerning problems in a general way—unconnected with your school work, as we may say; in other words, looking at the aerodynamic field in its broad plane. Is that right? A. Yes.

"Q. Now, tell me something, Mr. Chin: At any time while Garbell was at the school during that two-year period, did you ever hear him discuss, or did he ever tell you anything about, a wing design which he had conceived, which had good stalling characteristics and consisted of a three-section wing, wherein the air foils were changed in some fashion, or any fashion, from what might be said to be the standard arrangements?

"A. He mentioned to me a certain principle that could apply to accomplish the same thing that I was trying to work out in order to get a different principle, which is not completely unorthodox, on which certain information were available already from NACA reports.

"Q. The information from the NACA reports which you just referred to was made available when?

"A. The NAC Reports were made available at all times because the school subscribes, and I myself, personally, subscribed to it, and those reports come in periodically.

"Q. The information you refer to dealt with what particular subject—calibration of air foils?

"A. No, it dealt with—it isn't covered by just one report, it is covered by several reports. One is on the effect of lift characteristics as a function of the Reynolds number.

"Q. The Reynolds number relates to air foils, doesn't it?

"A. It relates to air foils in this way: it has to do with scale effect or size of the air foil; the scale effect of an airplane as compared with that of a tested air foil.

"Q. At that time, did these NACA reports have worked out what you might say the family curve of the airfoils? A. Yes.

"Q. From what you have said, I understand that for some time you yourself had been attempting to work out some principle of wing design utilizing the information in these reports?

"A. Yes, that's right.

"Q. In connection with your work on that idea, had you considered changing the scale, or graduating, I may say, the scale of the air foils from one section of the wing to the other?

"A. The size, or length, of the air foil, of necessity, does change, because of the root length, due to the plane—the form of the wing, long at the root and shorter at the tip; but at no time did I try to change the camber of the air foil not related to the same family. If I started, let us say, at 23,000 c's, I retained 23,000 c's right to the tip. The only variation is on the width of the root.

"Q. I follow you. Under your plan, the corresponding points of the different air foils would be connected by straight lines?

"A. That is right. That is exactly what I am trying to do, to get away from complicated structure.

"Q. I suppose that, with this in your mind, it was a natural consequence that you eventually got into a discussion of these principles with Dr. Garbell?

"A. Yes. In fact, I had worked out the data already, showing that if I twist the wing tip, using the same family air foils, 3 degrees, I would have moved the stall inboard, which is a conventional method, using this NACA information which I have just mentioned, because it was a function of the Reynolds number.

"Q. When you mentioned this to Garbell, of course, that sort of opened the door for a discussion of these matters? A. That's right.

"Q. And at that time did he say that he had worked out the principles to be used in a wing design?

"A. Yes. He mentioned in this broad sense, in the way of conversation, that the same thing I was trying to accomplish could be done by a different method which he had worked on before. But I have not seen any detail of the work, although he mentioned that the end result could be accomplished by a different principle.

"Q. Do you recall at any time, in conversations with you, or in any lecture, or anything like that, that Dr. Garbell referred to a three-section wing utilizing these principles?

"A. I do not recall the number of sections, but he mentioned to me that it could be done by switching sections—that means switching the family relation of the air foil; but I do not recall how many sections it required, how many switches it required.

"Q. You do recall he mentioned that this same thing could be done in more than one section on a wing? A. That is right.

"Q. And by switching the family curve of one section as distinguished from another, the two sections, or as many sections as there were, would differ, one from the other? A. Yes.

"Q. Now, the purpose of utilizing that principle, I take it, would be to move the stalling point of the wing inboard from the tip?

"A. That's right.

"Q. Particularly away from the ailerons?

"A. You don't have to move all the way inboard; and the ailerons generally covering the tip point of the wing—moving them inboard so as to permit a certain degree of control over the ailerons even during stalling.

"Q. You do recall the doctor saying that he had already worked that principle out?

"A. I have not seen any detail of the work, but he did say that he had worked on that particular principle. Let us put it that way.

"Q. Did you ever learn that prior to that time he had utilized this principle in connection with the construction of gliders that had flown? Do you recall that?

"A. I don't recall that he had built one—whether he did say that he had built one—but in other conversations he mentioned that he had built gliders before, his being a captain of the Italian Olympic Glider Team, or something like that; but I don't recall definitely whether he had actually used this particular principle in any of the gliders he may have built; but I do recall, in many other conversations, that he had built gliders before; whether he had applied that principle or not, I do not know, because when he mentioned this particular switching of wings to me, I agreed with him at that time, offhand it sounded all right, but my comment at that time was that probably it would give considerable structural difficulty in not having to pass straight lines between corresponding points on the wings, and complications would arise, wing jigs, and things of that sort. It is a mechanical problem, an aerodynamic problem.

"Q. Of course, such a wing, being built in sections, with different family curves in connection with the air foils, would present, I might say, a broken-line appearance of the completed wing, as distinguished from a straight-line appearance, from fuselage to wing tip. Is that correct?

"A. Yes. You see, I taught descriptive geometry and drafting, along with the design course, and anything complicated like that I would immediately see a structural or mechanical problem that would be difficult to overcome, so I did not pursue it any further, with the discussion we had, nor did I even try it myself, because of the mechanical difficulty that I would see.

"Q. Nothing was worked out? A. Yes.

"Q. In these discussions, Mr. Chin, where you and Dr. Garbell were conversing about this particular wing, would you say that those conversations were had in 1940?

"A. Yes. I would say during 1940—about that time.

"Q. Did they occur upon more than one time, or was the subject referred to now and then——

"A. It was never a continuous discussion, you might put it that way. Oh, maybe one or two other discussions after that. But I did recall this one particular time, where I had just completed my study of using the NACA data, at which time we discussed it, you might say, after hours, an hour; maybe a couple of times afterwards, maybe get ten or fifteen minutes of general discussions; but I don't recall that we pursued that discussion much further, because we had other problems to discuss.

"Q. At any time during these discussions, when the subject was mentioned, did Garbell use any figures or refer to any formula in connection with this principle? A. I do not recall.

"Q. You have no such recollection?

"A. I have no recollection on that, although, quite naturally, I talked with a pencil and paper a lot of the time and he talked with a pencil and paper a lot of the time; but I don't recall that he drew out any particular formula or——

"Q. Do you think during these discussions both of you or one of you drew sketches? Would you say that took place or did not take place?

"A. I would say from my own habits that it probably took place, but I do not recall what we drew.

"Q. Now, Mr. Chin, do you believe that you could state to me, in all fairness, from what you heard Dr. Garbell say, that at that time he had conceived and worked out this principle or a principle of wing design which could be applied to more than one section, so that there would be a dissimilarity of family curve of air foils between one section and the other?

"A. I would say, from the impressions that I had at that time, that he had conceived the idea; but I have no knowledge that he had worked it out completely, because we did not pursue it in any detail, only on the surface——

"Q. Now, I am going to put a question to you somewhat in the nature of considering you as an expert here. Suppose I would say to you that those principles had been used in the construction of a wing placed upon a glider and that the glider had successfully flown. Under those circumstances, would you believe that the principle had been worked out?

"A. Yes, you might say that would be the test or proof.

"Q. As to whether the thing had been worked out or not? A. Yes."

Upon diverse occasions prior to the commencement of the trial of this action affiant requested the said Harry B. Chin to testify upon the trial of said action on behalf of plaintiffs as to the subject matter contained in Mr. Chin's statement as hereinabove set forth. At all times Mr. Chin refused to testify.

Further affiant sayeth not.

/s/ THEODORE ROCHE, JR.

Subscribed and Sworn to before me this 31st day of January, 1951.

[Seal] /s/ FRANCES L. RICHMOND,
Notary Public in and for
Said County and State.

Comm. expires Mar. 7, 1954.

[Endorsed]: Filed February 5, 1951, U.S.D.C.

[Title of District Court and Cause.]

AFFIDAVIT OF MAURICE A. GARBELL

State of California,
County of Los Angeles—ss.

Maurice A. Garbell, being first duly sworn, deposes and says:

That he is the Maurice A. Garbell who has previously testified in the above-entitled cause and that if called to testify further would state as follows:

That he has read the reports referred to in the affidavits of Harry C. Matteson and William W. Fox and the conclusions reached by these men as to what the reports of California Institute of Technology, Galcit Report 504C, dated April 11, 1947, and C.V.A.C. Report ZA-240-008 and C.V.A.C. Flight Test Report of Flight No. 7 of the Model 110 airplane of August 19, 1946, show.

Further, affiant states that these reports do not show the Convair 240 airplance as certificated and sold; that alterations of the nacelles, wing fillets, ailerons and flaps were made to the airplanes as certificated and sold in order to permit the wing to stall as described in the patent in suit, and that in the airplane as certificated and sold the stall inception is over a large inboard area and that this stall progresses inboardward toward the root of the wing and that the stall of said airplane is not a root stall such as I defined a root stall in my testimony.

That Exhibit 35 discloses that after the alterations to the nacelles, the wing fillets, the flaps and the ailerons, the airplane stalled as I have described.

Further, affiant states that the Convair 240 as certificated and sold were airplanes that had modifications made to them to correct the stalling characteristics described in Flight Test Reports No. 6 and No. 7 of Exhibit 35.

/s/ MAURICE A. GARBELL.

Subscribed and Sworn to before me this 30th day of January, 1951.

/s/ IRENE J. KNUDSEN,
Notary Public in and for said County and State above written.

[Endorsed]: Filed February 5, 1951, U.S.D.C.

[Endorsed]: Filed August 25, 1951, U.S.C.A.

[Title of Court of Appeals and Cause.]

STIPULATION AND ORDER

It Is Hereby Stipulated by and between appellants and appellees under the provisions of Rule 76(h) that the record on appeal may be supplemented to include the following material omitted from the record on appeal:

(a) The translation marked as "Exhibit 32a" attached to Defendants' Exhibit AAA; and

(b) This stipulation.

And that this stipulation constitute a designation of the supplemental record to be printed as a supplement to the record heretofore filed in this cause and that the attached constitutes a true copy of said translation "Exhibit 32a" of Defendants' Exhibit AAA hereinabove identified and that the supplemental record so designated by this stipulation may be printed and will constitute a supplement to the record on appeal.

This stipulation is entered into at the request of appellants, and it is further stipulated that the cost of printing the supplement referred to herein shall be borne by appellant.

Dated August 24, 1951.

HARRIS, KIECH, FOSTER & HARRIS,

By /s/ FORD HARRIS, JR.,
Attorneys for Appellants.

LYON & LYON,

By /s/ LEWIS E. LYON,
Attorneys for Appellees.

So Ordered:

/s/ WILLIAM DENMAN,
Chief Judge,

/s/ CLIFTON MATHEWS,
Circuit Judge.

EXHIBIT No. 32a

Translation of Page 419, No. 16

"Flugsport"—1937

Performance Glider FS 16 "Wippsterz"

This plane was designed and built by the "Study-Group for Technology of Airplanes" at the "University for Technology" in Stuttgart. It made its first public appearance when crossing the Alps from Salzburg.

The aim, the designer had in mind, was to obtain high speed and maneuverability.

The cantilever high-wing is in two sections and is trapezoidal; the profiles from root to tip are: NACA 2318, 2315 and 4312. Considerable security against droop has been accomplished by root fairing. This plane can easily be kept in a straight direction by the use of the rudder, even if the elevator is "pulled." The ailerons are rather large and are made of dural; they have "levelling or compensation" tabs.

The fuselage is pulled up and backward, an idea which has proved itself with the "Fledermaus," particularly in bad terrain. Cantilever empennage, both rudder and stabilizer unbraced.

[Endorsed]: Filed August 27, 1951.

[Title of Court of Appeals and Cause.]

STIPULATION RE APPEAL RECORD

It Is Hereby Stipulated by and between the parties to the above-entitled appeal, through their respective attorneys, that the following exhibits and portions of exhibits originally designated for printing but omitted by the printer shall be printed in a supplement to the printed record on appeal.

Defendants' Exhibit A (Report on Airfoil Selection for the Revised Two-Engine Tailless Design ZA-101), pages 1 to 60, inclusive;

Defendants' Exhibit EE (Glen L. Martin Co. Engineering Report No. 1326);

Defendants' Exhibit OOO, last two (2) pages only;

It Is Further Stipulated that the following exhibits, previously designated for printing, need not be printed but may be considered by the Court in their original form without the necessity of reproduction:

Plaintiffs' Exhibit 35;
Defendants Exhibit LL;
Defendants' Exhibit PPP;
Defendants' Exhibit XXX.

Dated September 11, 1951.

HARRIS, KIECH, FOSTER & HARRIS,

By /s/ FORD HARRIS, JR.,
Attorneys for Apellants.

LYON & LYON,

By /s/ FREDERICK W. LYON,
Attorneys for Appellees.

Approved and It Is So Ordered.

.........................,
United States Circuit Judge.

[Endorsed]: Filed September 13, 1951.